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HIGH PERFORMANCE APOLLO PROPULSION SYSTEM STUDY

(UNCLASSIFIED TITLE)

NASA CONTRACT NAS 9-1729
FINAL REPORT
MARCH 1963

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VOLUME II

PROPELLANT SURVEY

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VOLUME II

HIGH PERFORMANCE APOLLO PROPULSION SYSTEM STUDY PROPELLANT SURVEY

March 1964
Contract NAS 9-1729

Prepared for
National Aeronautics and Space Administration

ROCKETDYNE
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FOREWORD

This report was prepared in compliance with the requirements for the National Aeronautics and Space Administration Contract, NAS 9-1729, "High Performance Apollo Propulsion System Study". The NASA technical monitors have been Mr. W. F. Eichelman and Mr. R. Brock at the NASA Manned Space Flight Center, and Mr. R. Rollins at the NASA Headquarters.

ABSTRACT

(Unclassified Abstract)

The results of the National Aeronautics and Space Administration Contract, NAS 9-1729, "High Performance Apollo Propulsion System Study" are presented in this report, Rocketdyne Report R5446. The report is composed of four volumes. This volume contains the analyses and results of the Propellant Survey Task. An extensive listing of propellant candidates for high performance Apollo propulsion systems was surveyed. The propellant combinations were numerically rated according to their relative merit in an Apollo application. Based on this rating, four candidate propellant combinations, for which propulsion systems could be operational by 1970, and six candidate combinations, for an operational date of 1975, were selected for further analysis in the next phase of the study. These analyses are contained in Volumes III and IV of this report.

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INTRODUCTION

Results of the Propellant Survey Task of NASA Contract, NAS 9-1729, "High Performance Apollo Propulsion System Study" are presented in this report. The purpose of this contract was to evaluate the use of high-energy propellants and advanced propulsion-system concepts to increase the landed-payload capability of the Apollo vehicle. The program was divided into two phases. In Phase I of this program, propellants and propulsion systems that will be operational by 1970 were considered, while in Phase II, systems were considered for a 1975 operational date.

Each phase of the program was composed of five tasks: I. Propellant Survey--review of propellants and candidate propellant selection, II. Propellant Selection--analysis of propulsion systems using the candidate propellants, III. System Design--vehicle and propulsion system design for a selected propellant combination, IV. Reliability Analysis--reliability analysis of the propulsion system design, V. Development Requirements--description of the development requirements necessary to realize the operational systems.

The purpose of the Propellant Survey Task was to establish the potential propellant candidates for the advanced Apollo propulsion systems. The 1970 (Phase I) and 1975 (Phase II) Propellant Survey results are both presented in this report. Four candidates were selected which could be developed into operational systems by 1970 and six candidates were selected for 1970 operational systems. In the survey the performance, physical characteristics, and availability of a large number of propellants were considered, and in a general manner, their effect on the propulsion system design and operation was indicated.

With the cognizance of the NASA, certain ground rules were established for the propellant survey (Table 1). Only propellants presently in existence were considered for both the 1970 and 1975 phases. There is considerable speculation about possible propellants which have not been synthesized; however, these propellants are hypothetical and would be impossible to analyze. In addition the propellants included in the survey must provide an increase in the landed payload. Therefore, propellant combinations which have a specific impulse and bulk density lower than the present combination, N_2O_4/50 percent N_2H_4-50 percent UDMH (N_2O_4/50-50), were immediately rejected. Propellants which have combinations of specific impulse and bulk density which result in less landed payload than the N_2O_4/50-50 combination were also rejected.

TABLE I

PROPELLANT SURVEY GROUND RULES

- Consider Synthesized Propellants Only
- Propellants Must Provide an Increase in Payload
- 1970 Systems

  1. Emphasize Minimum Design Changes
  2. Current NAA Service Module Design Concept
  3. Current Grumman LEM Design Concept
  4. Consider Only Liquid Propellants

- 1975 Systems

  1. No Design Change Limitations
  2. Life Support Capsules are Fixed
  3. Solid, Hybrid and Slurry Propellants May be Considered

To ensure that the 1970 systems will be operational by that date, a minimum design-change approach was adopted for these systems. The actual propulsion system changed as new propellants were used, but the basic Service Module and LEM vehicle structures were maintained. As a consequence of this minimum vehicle-change approach, only liquid propellants were considered for the 1970 operational date.

The systems to be operational in 1975 have no design restrictions. Only the life support capsules and the booster vehicle were maintained as currently designed. The lack of design restriction resulted from the long development period available. Solid, hybrid, slurry, and powder-type propellants were considered.

The propellant survey was approached with a complete, comprehensive listing of propellants based upon their chemical family. Included in the listing were bipropellant liquids, liquid mixtures, hybrids, metallic additives, and solids. A flow chart of the propellant survey is presented in Fig. 1. The position and effect of some of the ground rules can be seen. Based on the listing of propellants, a screening was made to eliminate the hypothetical propellants and to select for further study the propellants representing the best performance and physical properties from each chemical family. Scrutiny of the specific impulse and bulk density of the various combinations served to eliminate those with poor performance, while propellant combinations which provide a payload increase over the N_2O_4/50-50 combination were retained.

The remaining propellant combinations were then separated into the 1970 and 1975 categories based upon the physical state in which they are used. Liquid propellants were considered in both the 1970 and 1975 phases; while gels, slurrys, solids, hybrids and powdered propellants were considered only in 1975. In the 1970 evaluation, the propellant volume was screened and liquid propellant combinations with volumes significantly larger than the present propellant volume were assigned to the 1975 phase.

A numerical rating system was developed for further evaluation and comparison of the propellant combinations. In the rating system, performance (landed-payload increase), reliability, system operation, development ease, and launch-operation ease were the main rating areas. In each area, the factors involved were determined. Analytical-rating expressions were developed for each factor and the propellant candidates were compared to provide a relative rating. Each of the factors and the main rating areas were then weighted to provide the overall numerical rating.

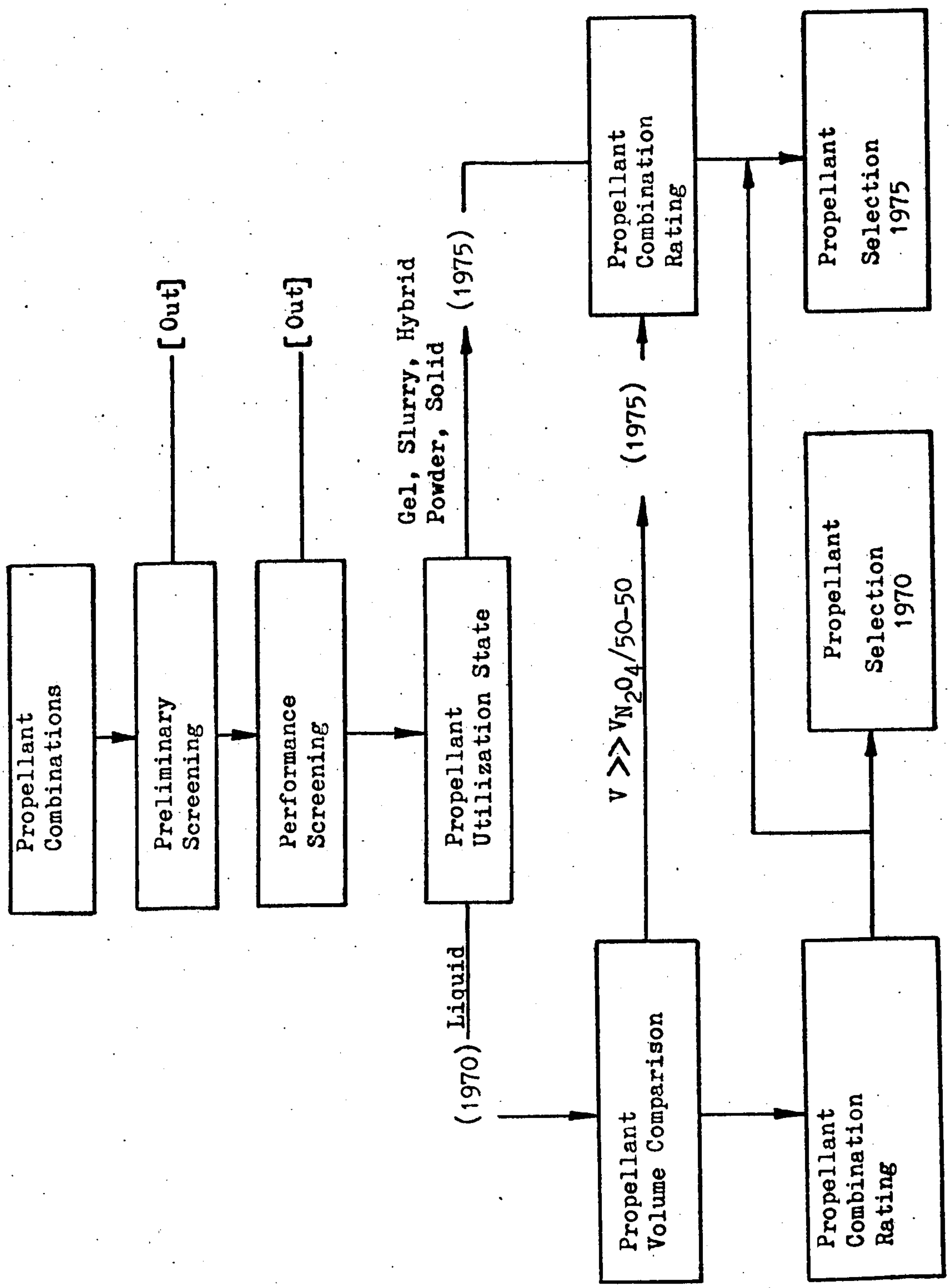

Fig. 1. Propellant Survey Flow Chart

Using evaluation factors that were developed, the propellant combinations were rated numerically in both the 1970 and 1975 categories. This numerical-comparison approach to the propellant survey has three outstanding features (1) The propellant listing ensures a comprehensive consideration of propellants in which no significant propellant will be neglected, (2) The evaluation and comparison of the propellant is systematic which facilitates a rational comparison, and (3) The importance of each factor resulting in a given numerical rating can be immediately determined and the factor isolated for review.

SUMMARY

Propellant combinations were surveyed for application in a high-performance Apollo vehicle which will provide an increase in landed payload over the present system. The propellant combinations were numerically rated according to their merit in this application. Four candidate propellant combinations were selected which could be operational by 1970 and six candidates were selected for a 1975 operational date.

The results of the propellant survey indicates that there are a number of high-performance propellant combinations that are well suited for use in the Apollo vehicle. Based upon the numerical rating, use of these propellant combinations will result in a considerable increase in the landed payload, and the propellants have the characteristics which permit their application to the Apollo propulsion system.

The overall numerical ratings for the 1970 propellant combinations are predominately oxidizer oriented. All of the high ranking combinations use fluorine-type oxidizers. The higher ratings are achieved by the F_2, FLOX (90 percent F_2, 10 percent O_2), N_2F_4, and OF_2 oxidizers in that order. In these overall ratings, F_2/N_2H_4 ranks the highest and the F_2 oxidizer combinations in general occupy the highest ranking positions. All of the top-ranking combinations have one cryogenic propellant. The top-ranking combination that is noncryogenic is Comp A/N_2H_4. (The hydrogen-fueled propellant combinations were excluded from the 1970 listings because of their low density and minimum design-change restriction).

To enable the Task II investigation to provide a distinctive propellant comparison with a broad scope of propulsion system configurations, candidate propellant combinations having different characteristics were selected. This selection will ensure that should undesirable features of a given propellant (oxidizer or fuel) be uncovered, all of the candidates will not be affected and the analyses can proceed without interruption. Four high ranking oxidizers were chosen: F_2, FLOX (90 percent F_2, 10 percent O_2), OF_2, and Comp A. The Comp A oxidizer was included as the highest ranking noncryogenic "earth storable" oxidizer. Fuels which give the best rankings with each of these oxidizers were then selected. The selections are given in Table 2 . Multiple fuels are indicated for the F_2 and OF_2.

These fuels are all high ranking and provide some flexibility in the thrust chamber cooling analyses.

TABLE 2

1970 PROPELLANT COMBINATION CANDIDATES

F_2/N_2H_4; NH_3

OF_2/B_2H_6; MMH; CH_4

FLOX (90)/MMH

Comp A/N_2H_4

In the selection of propellant combination candidates for the 1975 operational dates, there were two objectives. First, the 1975 propellant combination candidates must provide a payload capability comparable to the 1970 candidate propellants. Second, it was desired that representatives of the various propellant physical states be included regardless of their position in the overall ranking. With these objectives in mind, the 1975 propellant combinations were rated in a manner similar to the 1970 ratings. The ratings are somewhat less oxidizer-oriented because of some of the high performance fuels available for 1975 although fluorine-type oxidizers still predominate. The use of hydrogen, which was not considered in the 1970 evaluation because of its low density, and the use of some of the metallic additives shift the overall propellant combination ratings from being predominantly oxidizer oriented.

In making selections for the 1975 candidates, only the liquids, hybrids, and solid additive propellants were considered. Although the solid propellants listed could be of interest, there is the area of solid propellant start and cutoff technology that must be developed before they can be considered.

The 1975 selected propellant combinations are shown in Table 3. Six propellant combination candidates were selected: three bipropellant combinations, two metallic additive combinations and one hybrid combination.

TABLE 3

1975 PROPELLANT COMBINATION CANDIDATES

Selected 1970 bipropellant

F_2/H_2

$OF_2/C_2H_5B_{10}H_{13}$

$F_2/Be\ H_2$

$O_2/H_2 + Be$

$F_2/MMH + BeH_2$

These candidates include two of the high-rated liquid bipropellant combinations and the best combination from 1970. The high-performance 1970 combinations also rate very high for 1975. The selected 1970 combination is included with the 1975 systems, since advanced propulsion concepts will be considered for these systems. Also included in the selection are the highest-rated hybrid and two of the highest-rated metallic-additive propellant combinations.

PROPELLANT LISTING

PROPELLANT SPECTRUM

The liquid propellant survey was initiated with a listing of the complete liquid propellant spectrum in a logical sequence. From this spectrum of fuels and oxidizers, candidate propellants were selected. This complete listing was accomplished by classifying all chemical compounds, that might represent potential propellants, with respect to their most representative chemical species or families. The resulting chemical families are shown in Table 4 . This approach ensured the development of a comprehensive listing which virtually eliminated the possibility of overlooking a candidate fuel or oxidizer.

Each chemical family has been listed as either a fuel or oxidizer, although it was recognized that in some potential systems a particular compound might qualify as the opposite from which it is listed. In general, the fuels were classified in families representing the amine and CN groups, hydrazines, hydrocarbons, metallics, hydrogen, and mixtures; while the oxidizer groups were listed as either halogens, oxygens, nitrogen oxides, oxygen fluorides, nitrogen fluorides, NOF groups, or as mixtures. The listing of propellants is given in Appendix A.

No attempt was made to separate or distinguish between earth-ambient, normal liquids or gases, cryogenic, semicryogenic, and noncryogenic (semistorable liquids) in this preliminary listing. Although this propellant spectrum primarily represented liquids (melting point below earth-ambient), solids which have potential applications were also listed. Many such solids can be utilized in mixtures, slurries, or in dense-phase (solids fluidization) applications. However, only in the metallic fuels family and in the actual solid propellant components were distinctions made between solids or liquids.

Because two separate operational periods, 1970 and 1975, are presented and many compounds that are unknown or relatively undeveloped at the present could become operational as propellants in the intervening time period, each chemical family was divided into three classifications, hypothetical, laboratory characterization, and engineering characterization, which denoted the development stage of the chemical as a propellant. Those compounds, which have been hypothesized as potential propellants, but have not been synthesized, separated, or formulated, or are incompatible in mixtures, are classed as hypothetical. These compounds represent the present and planned future, analytical and experimental efforts of the propellant chemists. As discussed previously, these compounds were listed as a matter of interest and were not considered further. Those compounds which have recently been synthesized, but have not been extensively

TABLE 4

PROPELLANT FAMILY LISTING

CHEMICAL FAMILIES: FUELS

1. Amine; CN Family
2. Hydrazines and Substituted Hydrazine Family
3. Hydrocarbon Family
4. Metallic Compound Family
5. Hydrogen
6. Mixtures and Slurrys

CHEMICAL FAMILIES: OXIDIZERS

1. Halogen and Interhalogen Family
2. Oxygen, Peroxide and Trioxide Family
3. Nitrogen - Oxygen Family
4. Oxygen - Fluoride Family
5. Nitrogen - Fluoride Family
6. ONF Family
7. Mixtures

SOLID PROPELLANT COMPONENTS

1. Oxidizers
2. Fuels
3. Polymeric Binders
4. Oxidative Plasticizer
5. Solid Additive

characterized with respect to their potential as a propellant are placed in the laboratory-characterization stage. The final classification, which represents the compounds that have been taken from the laboratory and have been through, or are undergoing, characterization as a propellant, is shown as the engineering characterization stage. The solid propellant constituents are similarly classified.

Although it has been shown, or is fairly obvious to the propellant engineer, that some of the compounds and elements represented herein could not meet the performance goals of this program, these chemicals were listed nevertheless to provide a complete listing of potential propellants. There are undoubtably other members of the various families omitted from the listing. A listing including all chemicals would prove to be highly unwieldy and would not enhance the result of the program. The selection to this propellant listing was made from consideration of the element or compound as a propellant.

PRELIMINARY LIQUID PROPELLANT SELECTIONS

To maintain the entire program within the level of the effort and schedule assigned, it was necessary to reduce this complete liquid propellant listing to a comparatively few propellants for the complete evaluation of their applicability to the advanced Apollo system. Therefore, a preliminary selection of propellants from each chemical family was made. This preliminary selection was made from a review of the propellants in each family considering the availability, and potential performance and operational features. In this selection, the first restriction was the rejection of the hypothetical propellants. From the remaining propellants, the most attractive members from each family were selected based on the theoretical performance and the range of physical properties. The selections were guided in part by previous analyses of many of the propellants which identified the physical and performance features that are the most suitable for the present application.

Where applicable, at least one member of each family was selected. Although it was desirable to illustrate a range of physical properties within each particular family with the selection of a few members of that family, this was not feasible in some situations. Performance deficiencies (below that of the present Apollo system) limited physical ranges, undesirable physico-chemical characteristics, etc., minimized the selections in certain families.

In addition, this listing was reviewed to ensure that propellants which are of current interest were included. Specific attention was, also, directed to the inclusion of propellant combinations which represent all of the various physical states, e.g., hybrids, solids, slurrys, and liquids.

The selections were made with respect to the two time periods, 1970 and 1975. The assignment of each of the propellants to one of these two time periods was made on the basis of the probability of the necessary technology being available for the use of the propellant at that time. This assignment was based only upon the propellant technology and in certain situations conflicted with the propulsion-system study (i.e., The adaptation of the propellant system available for the 1970 period to the Apollo propulsion involved such a radical change in the configuration that the system will not be applicable until 1975) which determined the final selection.

The various fuel and oxidizer families, and the selected propellants from these families are discussed in the following pages.

Fuel Families

Amine and CN Family. The amine and CN family consists of a large number of well-characterized compounds with a wide range of physical properties. However, very few of these compounds are attractive as propellants from a performance standpoint. For the most part, this family has been considered only in mixtures with other compounds to improve or tailor the physical properties of a more attractive performance species. With the exception of ammonia (NH_3) which was selected from this family, no member of this family met the performance goal of the program.

Hydrazine Family. The hydrazine family has undergone extensive characterization and evaluation as storable propellants in the past from this family, hydrazine, (N_2H_4), unsymmetrical dimethyl hydrazine (UDMH), monomethylhydrazine, (MMH), ($CH_3N_2H_3$) were selected. The selections represent the most suitable characteristics (for the considered application) found in this family of propellants. Essentially these three fuels achieve the level of performance of the present Apollo fuel and were selected for comparative purposes. However, these fuels do possess improved performances with certain oxidizers and metallic additives and appear to be more attractive from this, as well as the physical standpoint, than the present Apollo fuel.

Hydrocarbon Family. The hydrocarbon family consists of a large number of compounds with a variety of engineering properties, which are fairly well characterized and comparatively inexpensive and available. Selected oxidizers dominate the potential performance levels of these compounds. Methane (CH_4), ethane (C_2H_6), and RP-1 were selected from this fuel family. These candidates achieve performance consistent with the family and embody liquidus ranges from the earth-ambient semicryogenics to the noncryogenics (storables). The compounds in this family represent an area in which a final selection of a compound from this family depended on a desired range of engineering properties rather than a wide difference in the performance level.

Metallics Family. The compounds consisting of the various metallics and their hydrides are combined in one group to form the metallics family. This group of compounds was divided into areas of one and two-component liquids and solids in addition to the development status subdivision. It was within this area that the greatest potential gains from fuels were recognized. There was a variety of both liquid and solid metallics, and metallic hydrides that indicated significant gains with selected oxidizers. Some of these compounds were available for the 1970 time period, but the majority of this family of chemicals was placed in the 1975 time period. Many of the elements and compounds selected from this group were solids adaptable for liquid systems by means of soluble mixtures, slurries, dense phase fluidization, etc. For the most part, such applications were discussed under mixtures.

Diborane (B_2H_6) and pentaborane (B_5H_9) were selected to represent the one component metallic liquids available for the use in the 1970 period and to provide two different liquidus ranges. Hybaline A_5, one of a series of amine adducts of aluminum borohydride undergoing present characterization studies, was selected to represent two-component metallic propellant available by 1970.

Several solid and solid-metallic propellants were selected: Al, AlH_3, Be, BeH_2, Li, LiH, and $(CH_2)_x$. The metallics can be used in a strict bipropellant application (a hybrid) or as an additive to a liquid bipropellant combination. Of these metallics, the BeH_2 is probably of greatest interest in that it provides large increases in performance. Because of the lack of application technology associated with these propellants, they were all placed in the 1975 development period.

Miscellaneous. Hydrogen, H_2, was placed in a category of miscellaneous fuels because it did not belong in any of the other categories, and was a logical selection for study.

Fuel Mixtures. A category of mixtures was used to represent miscible liquid mixtures of the same families and different families as well as soluble mixtures, slurries, and other similar applications of solid fuels with liquid fuels. Such mixtures represent a large segment of past, present, and future fuel systems. Many of these are systems designed to provide tailored engineering properties, such as the 50-50 hydrazine-unsymmetrical dimethlhydrazine, Hydrazoid P and Hydyne mixtures selected to represent presently-available mixtures that appear most suited to the present application.

50 - 50	50 percent N_2H_4 - 50 percent UDMH
Hydrazoid P	5 mole N_2H_4 - 4 mole MMH - 1 mole $HClO_4$
Hydyne	60 percent UDMH - 40 percent DETA (diethylenetriamine)

Oxidizer Families

Halogen and Interhalogen Family. From the halogen and interhalogen oxidizer family fluorine (F_2), chlorine trifluoride (ClF_3), chlorine pentafluoride (ClF_5) and bromine pentafluoride (BrF_5) were selected. The first two oxidizers are well-known dense liquids with widely separated liquidus ranges and performance levels. ClF_5 is a recently-synthesized family member, which possesses an attractive performance level and physical state that lies between those of F_2 and ClF_3. The BrF_5 is a familiar, low-performance propellant. All of these assigned to the 1970 period.

Oxygen, Peroxide, and Trioxide Family. Oxygen (O_2) and hydrogen peroxide (H_2O_2) were selected from this family. Oxygen, a cryogenic member of the family, has demonstrated good performance with a wide variety of fuels. The noncryogenic compound, H_2O_2, demonstrates nominal performance levels with all fuels except the majority of the metallics, with which it shows very high performance potential. Water (H_2O) was selected as a special representative because it possesses a high performance potential with a particular fuel, BeH_2, and represents an easily handled oxidizer system. However, because of the status of the fuel, this oxidizer has been assigned to the 1975 period.

Nitrogen Oxides and Nitro Family. The nitrogen oxide and nitro family were used to designate those groups whose chemical state were characterized by a nitrogen-oxygen species. The potential performance level of this entire group of compounds is represented by nitrogen tetroxide (N_2O_4). This oxidizer is the basic earth-ambient storable oxidizer in present use and was selected for comparative purposes. Some nitric acid (HNO_3) mixtures were considered under the oxidizer mixtures.

Oxygen Fluoride Family. The oxygen fluoride family, which is constituted by a number of compounds predominately affected by the presence of an OF group, consists of a number of highly reactive compounds with attractive performance levels. The presence of both oxygen and fluorine lends an almost universal application with all types of fuels. From this family oxygen difluoride (OF_2) was selected. Although OF_2 appears to represent the maximum performance level achievable by a member of this family, other recently synthesized members may offer more attractive engineering properties.

Nitrogen Fluoride Family. The nitrogen fluorides are a comparatively new family of compounds characterized by a NF species. The most suitable propellants from this family are tetrafluorohydrazine (N_2F_4) and nitrogen trifluoride (NF_3). Much synthesis effort has been applied in the area over the past few years without appreciable success. Efforts to duplicate performance levels near those of this family's selections with NF compounds of more desirable engineering properties were centered in the area of CNF compounds. Several hundred such compounds have been synthesized most of which are highly sensitive, unstable, and/or possess low performance levels. However, the use of these compounds in mixtures may permit utilization of their more attractive properties. Such mixtures are included in the oxidizer mixture selections.

Oxygen-Nitrogen Fluoride Family. The oxygen nitrogen fluoride family is a small family of compounds, which are essentially a specialized group of nitrogen fluorides. Although very few of these compounds have been synthesized, this family appears to have more potential usefulness than the CNF compounds with respect to less sensitivity and higher performance.

From this family, trifluoroamine oxide (NOF_3) and nitryl fluoride (NFO_2) were selected. A few other members that have been synthesized are being considered for use in mixtures.

Oxidizer Mixtures. As in the case of the fuels, several oxidizer mixtures were considered. The requirements for specified performance levels and certain physical characteristics have been met in many situations by the tailoring of a basic oxidizer with the addition of compounds reflecting the desired set of characteristics. Most of these mixtures have undergone sufficient development to be considered for 1970 application. The oxygen-fluorine (FLOX) and mixed oxides of nitrogen (MON) mixtures are selections embodying two vastly different areas of performance and engineering properties. The MON mixture represents a tailoring of the present Apollo oxidizer to obtain a lower liquidus range, while the O_2 - F_2 mixture represents the tailoring of an oxygen system to improve performance and density without changing desirable engineering properties. The ClF_3 - ClO_3F selection represents a high-density, earth-ambient noncryogenic oxidizer, which was considered for use with selected metallic hydrides (addition of oxygen through the introduction of ClO_3F increased the performance of the basic ClF_3 system). The selection of MOXIE-1 and MOXIE-2a represents the recently-considered mixtures of NF compounds which were formulated to reduce sensitivities and increase storability while maintaining a desirable performance level.

The two mixtures, maximum density fuming nitric acid (MDFNA) and inhibited red fuming nitric acid (IRFNA) were selected as candidates since they have the characteristics of fairly high density propellants. These are primarily mixtures of HNO_3 and N_2O_4.

FLOX(N)	N percent F_2 - (100-N) percent O_2
MON	75 percent N_2O_4 - 25 percent NO
MOXIE-1	N_2F_4 - ClF_3 - ClO_3F (no percentage compositions assigned)
MOXIE-2A	42 percent N_2F_4 - 42 percent CN_3F_7 - 15 percent ClO_3F
MDFNA	56 percent HNO_3 - 44 percent N_2O_4
IRFNA	85 percent HNO_3 - 15 percent NO_2

SOLID PROPELLANT COMPONENT SELECTION

The solid-propellant constituents listed in Appendix A were surveyed and attractive propellants selected. Selections were made to include all of the important propellant classes envisioned at the present time. In general, the fuels selected were the metallics and metallic hydrids. Oxidizers were representative of various oxidizer families.

There are some omissions, in particular the BN and Be_3N_2 systems. The substantial R&D effort on the solid BN systems has almost been abandoned completely because the realizable solid systems had low density, relatively low theoretical impulse (usually < 290), poor physical properties and probably poor combustion efficiencies, with certain exceptions such as hydrazine bisborane monopropellant. Similarly oxidizers such as lithium perchlorate or hydrazine perchlorate provide incremental gains in density or impulse, but these gains are not in general large enough to offset other problems of compatibility or sensitivity which they introduce.

SELECTED PROPELLANTS

The resulting list of propellants is presented in Table 5. This list presents the most suitable candidate propellants (fuels and oxidizers) from each chemical family. In the listing, it should be noted that the N_2H_4 listed as an oxidizer is considered to be in this category only when used with B_5H_9. The lower portion of the fuel listing contains a series of solid propellants which are used with various oxidizers as a hybrid system. Many of these same solids are listed as additives to be used with a liquid, bipropellant combination. The solid-propellant combinations and individual propellants were considered only for the 1975 period.

TABLE 5

PROPELLANTS CONSIDERED IN THE SURVEY

Oxidizers	Fuels	Additives**	Solid Oxidizers **
1. F_2	1. NH_3	1. Be	1. BeH_2/NH_4ClO_4
2. ClF_3	2. N_2H_4	2. Al	2. Be/NH_4ClO_4
3. Compound A	3. MMH	3. BeH_2	3. NO_2ClO_4
4. BrF_5	4. UDMH	4. AlH_3	4. $N_2H_6\ (ClO_4)_2$
5. ClO_3F	5. CH_4	5. Li	5. Difluoramino Oxidizers
6. O_2	6. C_2H_6	6. LiH	
7. H_2O_2 (98percent)	7. RP-1		
8. N_2O_4	8. B_2H_6		
9. HNO_3	9. B_5H_9		
10. OF_2	10. Hybaline A-5		
11. NF_3	11. Hybaline B-3		
12. N_2F_4	12. H_2		
13. ONF_3	13. N_2H_4/UDMH		
14. NFO_2	14. Hydrazoid P		
15. MON	15. Hydyne		
16. MOXIE 2A	16. $C_2H_5B_{10}H_{13}$		
17. MDFNA	17. AlH_3**		
18. IRFNA	18. BeH_2**		
19. ClF_3/ClO_3F	19. (CH_2)**		
20. FLOX	20. Li**		
21. N_2H_4*	21. LiH**		
	22. Be**		
	23. Al**		

* With B_5H_9

** 1975 Systems Only

PROPELLANT PHYSICAL PROPERTIES

To provide a background for comparing the various propellants, physical property data were assembled. These data are presented in Tables 6 to 9 . The following information is listed.

1. Propellant Density
2. Normal Boiling Point
3. Normal Freezing Point
4. Vapor Pressure vs Temperature
5. Liquid Specific Heat Capacity
6. Molecular Weight

In some cases, little propellant data existed and it was necessary to make estimates. This situation accurred primarily in the area of specific heat capacity and vapor pressure. Estimates of heat capacity were made using Kopp's rule and corrected by comparison to data for a similar propellant. For some mixtures, a weighted average of the individual propellants was used. Vapor pressure data were, in some cases, extrapolated from a few data points to other temperatures. The specific heat capacity is listed at either the normal boiling point or at 70 degrees F depending upon which is the lower temperature value.

TABLE 6

LIQUID OXIDIZER PHYSICA

Oxidizer	Normal Boiling Point F	Normal Freezing Point F	Specific Gravity at	Temp. F	Specific Heat Capacity BTU/LB F	Temp. F	Pressure psia
A	6.8	-153.4	1.899	NBP	0.31	N.B.P.	2.31
BrF_5	104.5	80.5	2.482	68	0.21	68	0.075
ClO_3F	-52.3	-231.0	1.710	NBP	0.227	N.B.P.	
ClF_3	53.15	-105.38	1.85	NBP	0.305	N.B.P.	17.2
FLOX (30-70)	-301	-362	1.232	-300	0.378	N.B.P.	8.5
FLOX (90-10)			1.46				
FNO_2	-81	-218	1.57	-150	.415	N.B.P.	
F_2	-307	-363	1.509	NBP	0.366	N.B.P.	0.035
HNO_3	181	-43	1.52	68	.423	68	2.3
H_2O_2 (98-percent)	299.2	27.5	1.432	77	0.635	N.B.P.	0.04
IRFNA	150.0	-57	1.57	68	0.41	68	
MDFNA	86	-35	1.528	68	0.429	68	
MON (85-15)	45	-45	1.40	45	0.382	N.B.P.	
MON (75-25)	21.5	-76	1.381	68	0.391	N.B.P.	
MOXIE 2A	-94		1.64	-60	0.422	N.B.P.	
NF_3	-199.2	-343.3	1.538	N.B.P.	0.244	N.B.P.	
NO	-241	-257	1.27	-241	0.46	N.B.P.	
N_2F_4	-99.4	-264	1.66	-99.4	0.51	N.B.P.	
N_2O_4	70.1	11.8	1.447	68	0.36	N.B.P.	2.9
OF_2	-228.64	-370.84	1.53	N.B.P.	0.281	N.B.P.	0.06
ONF_3	-125		1.9		0.402	N.B.P.	
O_2	-297.6	-361.8	1.14	N.B.P.	0.406	N.B.P.	7.3
O_3	-170	-316	1.33	N.B.P.	0.357	N.B.P.	1
RFNA	148	-56	1.55	68	0.419	68	1

L PROPERTIES

	Vapor	Pressure							
Temp. F	Pressure psia	Temp. F	Pressure psia	Temp. F	Pressure psia	Temp. F	Pressure psia	Temp. F	Molecular Weight
-65.2			50	58	63.9	68	268	158	130.5
-70	6.3	68	50	183					174.9
	12	-60	50	2	56	10	156	68	102.5
60	39.7	100	50	112	80.6	140			92.5
310	26.5	-290	50	-274	66.5	-270	140	-250	33.8
			50						37.2
			50	-35					65.0
363	0.1	-356	50	-280					38.0
100	10	160	50	248					63.0
77	10	280	50	370	100	412			34.0
	17.3	160	50	225					61.4
	10	70	50	150	100	195			73.2
			50	105					82.7
	30	40	50	67	93	100	288	160	76.5
			50	-42					139.1
	10	-220	50	-173	100	-156			71.0
			50	205					30.0
			50	-47					104.0
12	13.92	68	50	121	100	155			92.0
312.7	13.39	-234.4	50	-197					54.0
			50	-75					87.0
310	10	-300	50	-273	615	-190			32.0
225	10	-180	50	-133	100	-110			48.0
2	10	130	50	220	100	265			59.7

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LIQUID FUEL PH

Fuel	Normal Boiling Point, F	Normal Freezing Point, F	Specific Gravity at	Temp F	Specific Heat Capacity BTU/LB F	Temp F	Pressure psia
B_2H_6	-135	-265	0.450	N.B.P.	0.66	N.B.P.	
B_5H_9	140	-53	0.627	68	0.57	77	3
CH_4	-259	-297	0.440	N.B.P.	0.84	N.B.P.	3.6
C_2H_5OH	173	-174	0.785	68	0.58	68	1.0
C_2H_6	-127	-278	0.546		0.60	N.B.P.	
$C_3H_7NO_3$	231	-150	1.52	68	0.42	68	1.0
$C_{10}H_{20}$	344	-110	0.805	68	0.47	68	1.0
H_2	-423	-435	0.071	N.B.P.	2.23	N.B.P.	1.02
HYBALINE A5	505	058	0.736	68	0.62	68	0.06
HYDRAZOID -P	243	-150	1.095	77	0.64	N.B.P.	0.48
JPX	211	-71	0.764	68	0.59	68	1.00
(MAF-4)-HYDYNE	148	-120	0.859	68	0.65	75	2.64
MMH	188	-62	0.8765	68	0.70	N.B.P.	1.0
NH_3	-28	-108	0.68	N.B.P.	1.07	N.B.P.	
N_2H_4	236	35	1.008	68	0.74	68	0.20
N_2H_4(50-50)UDMH	158	19	0.8986	77	0.69	77	2.0
RP-1	422	-55	0.806	68	0.45	68	0.3
UDMH	146	-71	0.784	77	0.65	68	1.89
$C_2H_5B_{10}H_{13}$	500	-65	.82	77	0.50	68	0.1

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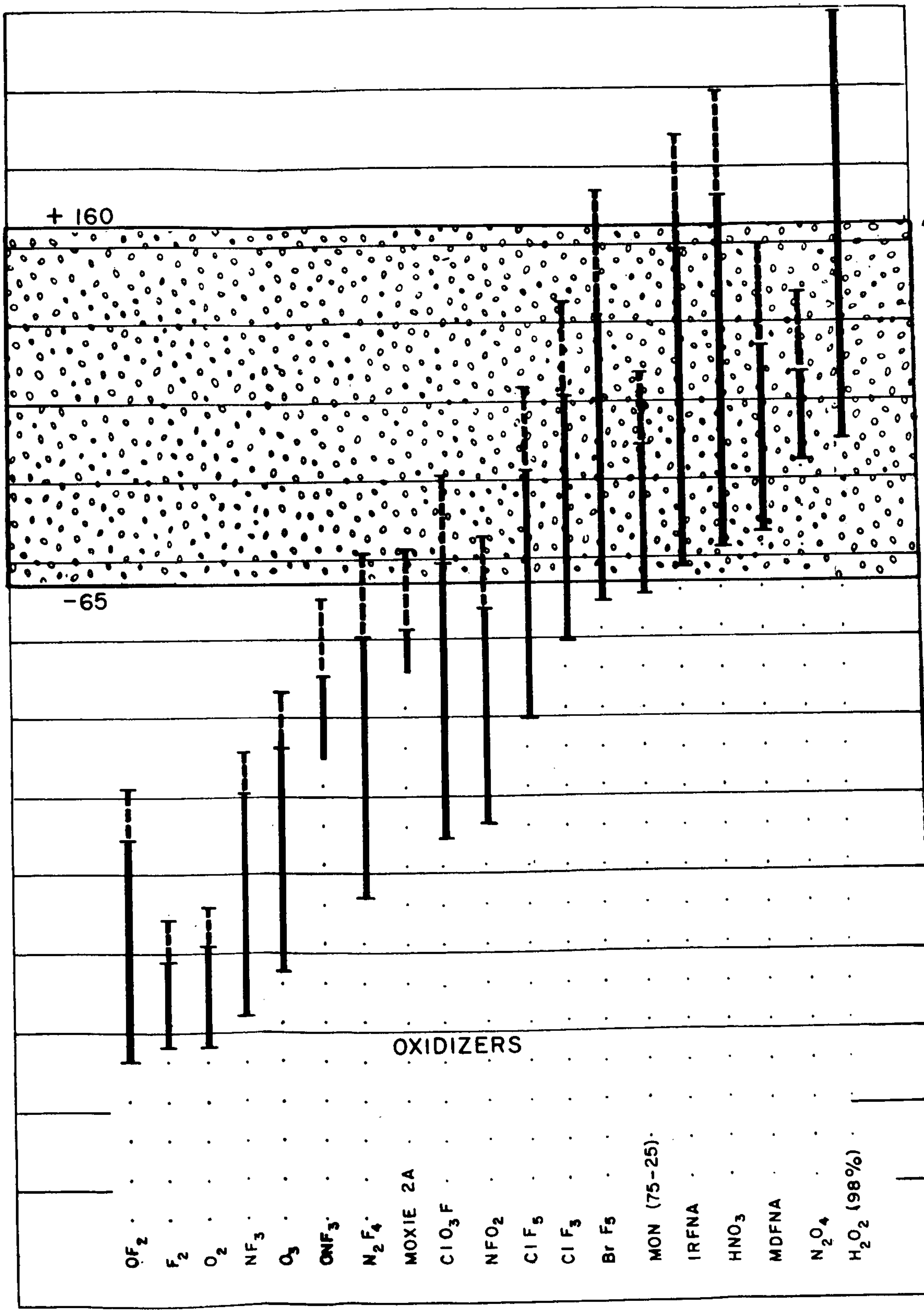
SICAL PROPERTIES

Vapor Pressure									
Temp F	Pressure psia	Temp F	Pressure psia	Temp F	Pressure psia	Temp F	Pressure psia	Temp F	Molecular Weight
	9	-150	-50	-90	59	-82	529	60	27.7
68	10	118	50	210	100	260			63.2
-285	32	-240	50	-225					16.0
73	10	155	50	235	100	280			46.1
			50	-78	544	70			30.1
100	10	205	50	315	100	375			105.1
190	10	320	50	450	100	520			140.3
-435	15	-423	50	-414	100	-407			2.0
77			50						134.3
67	0.99	103	50						44.4
90	6	160	50	286	100	342			89.3
68	18	160	50	219					72.1
77	10	160	50	245	100	310			46.1
	10	-40	50	22	60	30	29	70	17.0
68	10	215	50	310	100	360			32.0
68	15	160	50	235					41.8
160	10	395	50	530	100	605			172.0
60	10	120	50	222	100	260			60.1
68	14	500							

TABLE

NORMAL BOILING AND FREEZING P



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8

OINTS OF PROPELLANTS

TEMPERATURE AT 50 PSIA
NORMAL BOILING POINT
NORMAL FREEZING POINT

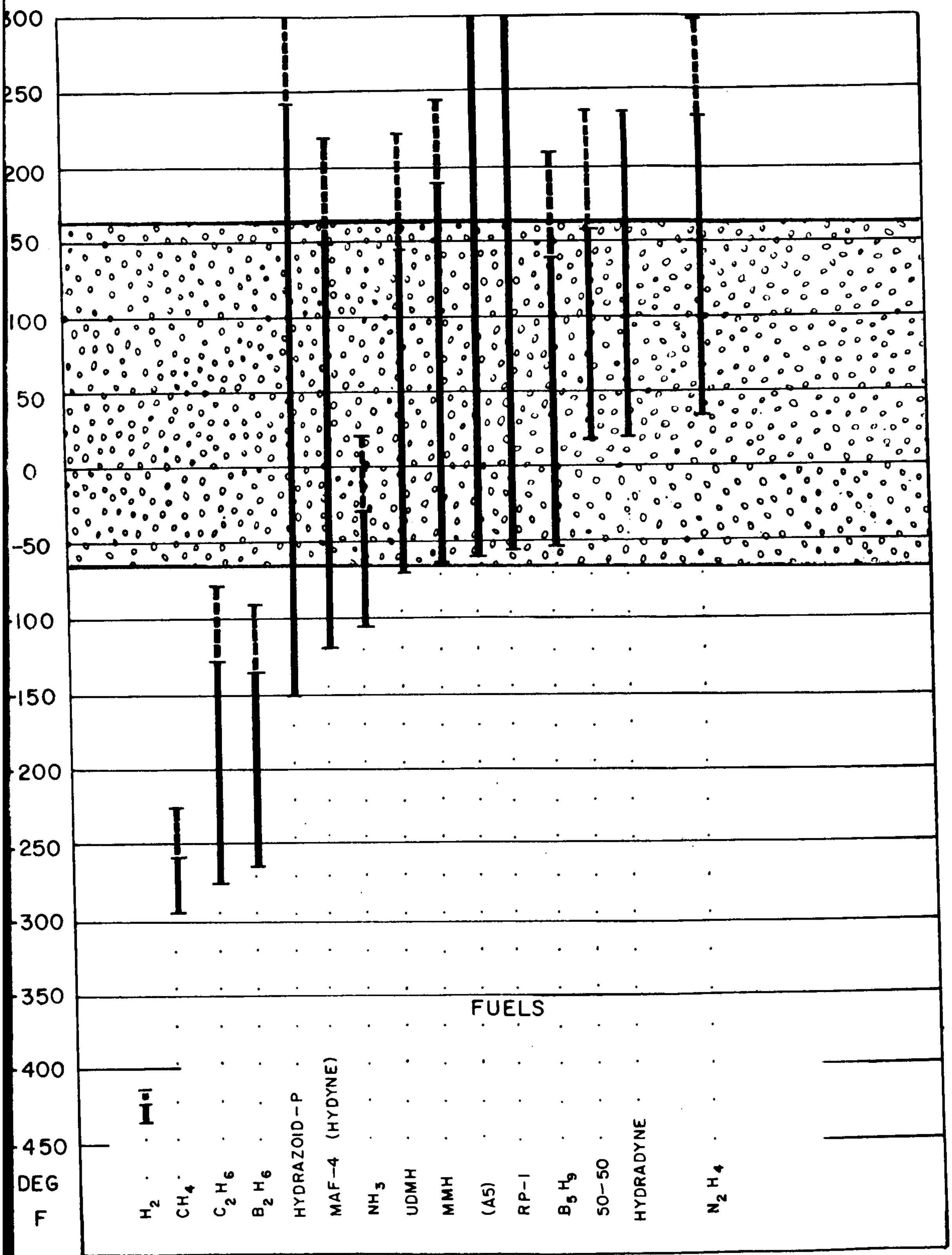

Table 9

METALLIC FUEL OR ADDITIVE PROPERTIES

Formula	Name	Specific Gravity	Temperature
Be	Beryllium	1.85	68
BeH_2	Beryllium Hydride	0.65	-
Al	Aluminum	2.70	68
AlH_3	Aluminum Hydride, alone	1.73	-
α Phase		1.49	-
β Phase		-	-
γ Phase		$\sim$1.3	-
Li	Lithium	0.534	68
LiH	Lithium Hydride	0.820	68
LiH H_4	Lithium Aluminum Hydride	0.917	77
B	Boron	2.5	-
$LiBH_4$	Lithium Borohydride	0.66	77
Li NO_3	Lithium Nitrate	2.38-2.40	-
Mg	Magnesium	1.74	68
Mg H_2	Magnesium Hydride	1.45$\pm$0.08	-

PROPELLANT PERFORMANCE SCREENING

One of the ground rules of the study was that only propellant combinations providing an increase in landed payload over the present N_2O_4/50-50 combination would be considered. The propellant performance screening was conducted to determine which propellant combinations resulted in a payload capability less than the N_2O_4/50-50 system. These combinations providing less payload were then eliminated as candidates.

Specific impulse calculations were made for various combinations of the propellants listed in Table 10. Specific impulse was determined for optimum expansion from 1000 psia to 14.7 psia, using the assumption of chemical equilibrium. Values of specific impulse are presented at the weight mixture ratio (oxidizer/fuel) which provides maximum specific impulse.

The essential performance data of the various liquid propellant combinations are listed in Table 10. These data were used throughout the propellant survey. In this table, the propellants are organized in alphabetical order by oxidizers and then by fuels. A number of other items characterizing the propellant combination are also listed in Table 10. Reading from left to right in the table, the following properties are listed: (1) Combustion temperature (degrees K), (2) oxidizer, (3) additive to oxidizer, (4) fuel, (5) additive to fuel, (6) specific impulse (seconds), (7) oxidizer/fuel weight mixture ratio (O/F), (8) bulk specific gravity, (9) performance assumption, and (10) overall propellant weight fractions. The performance-assumption column indicates instances where the performance calculation deviates from the usual calculations. Where there is no indication, the performance was calculated as related in the preceeding paragraph, using the latest heat-of-formation data. An "O" in this column indicates that only performance data based on an outdated heat of formation were available, while an "E" identifies an estimated specific-impulse value. The estimates were based upon calculations from similar propellant combinations and should be accurate to approximately 1-2 seconds of specific impulse. These situations were infrequent and did not occur for any of the major combinations. Some combinations with solid additives to the fuel show "zero" percent relative weight for either the fuel or the additive. This indicates that the highest performance is achieved by the liquid bi-propellant or by the hybrid combination.

TABLE 10a

PROPELLANT COMBINATION PERFORMANCE CHARACTERISTICS

Temp	Oxidizer	Fuel		I_S*	MR	S.G.	Rel. wts.
	BRF5	B5H9		246	11.45	1.99	92-00-08-00
	BRF5	DETA		220	3.68	1.84	78-00-22-00
	BRF5	MMH		236	3.60	1.78	78-00-22-00
	BRF5	N2H4		244	3.35	1.86	77-00-23-00
	BRF5	UDMH		235	3.60	1.77	78-00-22-00
4297	CLF3	ALH3		283.5	3.75	1.71	79-00-21-00
5096	CLF3	BE		288.	3.8	1.83	79-00-21-00
3920	CLF3	BEH2		315.4	3.7	1.33 0	79-00-21-00
C	CLF3	B10H13C2H5		280.	6.0	1.57	86-00-14-00
4195	CLF3	B2H6		297.3	7.0	1.33	87-00-23-00
4375	CLF3	B5H9		289.8	7.	1.47	88-00-12-00
4375	CLF3	B5H9	AL	289.8	7.	1.47	88-00-12-00
4375	CLF3	B5H9	ALH3	289.8	7.	1.47	88-00-12-00
4826	CLF3	B5H9	BE	292.6	4.55	1.55	82-00-13-05
3920	CLF3	B5H9	BEH2	315.4	3.7	1.33 0	79-00-00-21
3510	CLF3	CH4		273.0	5.66	1.22	85-00-15-00
3510	CLF3	CH4	AL	273.0	5.66	1.22	85-00-15-00
4297	CLF3	CH4	ALH3	283.5	3.76	1.71	79-00-00-21
5096	CLF3	CH4	BE	288.0	3.76	1.83	79-00-00-21
3918	CLF3	CH4	BEH2	315.4	3.54	1.33	78-00-00-22
	CLF3	CH4	LI	315.3	3.22	1.16	76-00-00-24

*Optimum sea level expansion, 1000 psia to 14.7 psia

TABLE 10a(CONT.)

Temp	Oxidizer		Fuel		I_S*	MR	S.G. E	Rel. wts.
	CLF3		C2H6		269.			
4116	CLF3		HYBAL AS		291.0	5.00	1.47	83-00-17-00
3895	CLF3		HYDRAZOID-P		290.1	2.3	1.52	70-00-30-00
C3597	CLF3		HYDYNE		275.	2.93	1.42	75-00-25-00
3422	CLF3		H2		318.0	11.72	0.62	92-00-08-00
	CLF3		LI		319.6	3.22	1.16	76-00-24-00
4109	CLF3		LIH		293.0	5.25	1.54	84-00-16-00
4465	CLF3		MGH2		275.0	3.55	1.74	78-00-22-00
3711	CLF3		MMH		286.0	2.70	1.42	73-00-27-00
3519	CLF3		NH3		275.0	3.65	1.34	79-00-21-00
3869	CLF3		NH3	AL	282.2	3.55	1.33	78-00-19-03
3860	CLF3		NH3	ALH3	289.7	3.22	1.41	76-00-14-10
4441	CLF3		NH3	BE	295.2	3.65	1.56	79-00-09-12
	CLF3		NH3	LI	314.5	5.0	1.36	83-00-16-01
3395	CLF3		N2H4		292	2.90	1.52	74-00-26-00
	CLF3		N2H4	AL	294.0	2.5	1.54	71-00-23-06
	CLF3		N2H4	BE	300.4	2.8	1.70	74-00-13-13
	CLF3		N2H4	MMH	265.	2.87	1.44	74-00-03-23
	CLF3		N2H4	N2H5NO3	286.7	1.6		62-00-23-15
3608	CLF3		N2H4	UDMH	284.3	2.90	1.45	74-00-13-13
C3506	CLF3		RP1		258.0	3.26	1.41	71-00-29-00
C3658	CLF3		UDMH		278.0	3.10	1.38	76-00-24-00
4425	CLF3	FCLO3	B10H13C2H5		286.3	7.0	1.43	79-08-13-00

*Optimum sea level expansion, 1000 psia to 14.7 psia

TABLE 10a(CONT.)

Temp	Oxidizer	Fuel	Fuel		I_S*	MR	S.G.	Rel. wts.
4297	CLF3	FCL03	B5H9		301.44	4.00	1.17	16-64-20-00
4474	CLF3	FCL03	B5H9		296.45	6.25	1.39	64-22-14-00
3675	CLF3	FCL03	N2H4	MMH	285.7	3.0	1.34	67-08-03-22
4545	CLF5		B10H13C2H5		299.	6.0	1.60	86-00-14-00
4503	CLF5		B2H6		317.0	7.0	1.36	87-00-23-00
3721	CLF5		CH4		293.2	3.0	1.04	75-00-25-00
4416	CLF5		HYBAL A5		309.3	5.00	1.51	83-00-17-00
4080	CLF5		HYDRAZOID-P		307.8	2.0	1.52	67-00-33-00
3792	CLF5		MMH		299.8	2.7	1.45	73-00-27-00
3390	CLF5		NH3		300.	3.8	1.27	79-00-21-00
4165	CLF5		N2H4		312.9	2.7	1.47	73-00-27-00
4579	CLF5		N2H4	BE	317.0	3.0	1.57	73-00-19-08
3866	CLF5		N2H4	UDMH	301.8	2.72	1.46	73-00-13-13
	CL03F		(BH4)2ALC2AL(BH4)2		295.2	2.40		71-00-29-00
	CL03F		AL(B3H8)3		300.5	3.20		76-00-24-00
4029	CL03F		ALH3		301.0	1.00	1.72	50-00-50-00
4241	CL03F		BE	B5H9	299.5	3.30	1.13	77-00-02-21
4173	CL03F		BEH2		339.0	2.13	1.12	68-00-32-00
3737	CL03F		B2H6		314.8	3.00	0.93	75-00-25-00
4466	CL03F		B5H9		316.3	4.00	1.14	80-00-20-00
B3610	CL03F		CH4		286.0	5.50	1.05	85-00-15-00
C3633	CL03F		HYDYNE		287.0	2.80	1.22	74-00-26-00
C2972	CL03F		H2		344.0	6.00	0.38	86-00-14-00

*Optimum sea level expansion, 1000 psia to 14.7 psia

TABLE 10a (CONT.)

Temp	Oxidizer	Fuel		I_s*	MR	S.G.	Rel. wts.
	CL03F	LIB4H9		303.0	3.40		77-00-23-00
3351	CL03F	LIH		272.0	3.17	1.36	76-00-24-00
3237	CL03F	MGH2		260.0	0.85	1.56	46-00-54-00
3611	CL03F	MMH		292.0	2.24	1.32	69-00-31-00
B3100	CL03F	NH3		273.0	2.02	1.05	67-00-33-00
3433	CL03F	N2H4		295.5	1.40	1.22	58-00-42-00
	CL03F	N2H4	N2H4.BC6H3	324.6	1.16		54-00-00-00
3602	CL03F	N2H4	MMH	292.4	2.20	1.33	69-00-04-27
	CL03F	N2H4.B3H7		308.9	2.30		70-00-30-00
C3686	CL03F	RP1		280.0	4.35	1.25	81-00-19-00
C3650	CL03F	UDMH		290.0	2.70	1.17	73-00-27-00
3900	FLOX (30-70)	B2H6		354	2.9	0.85 E	74-00-26-00
4450	FLOX (30-70)	B5H9		335	3.2	1.00 E	76-00-24-00
3800	FLOX (30-70)	HYBAL A-5		332.	1.7	0.99	63-00-37-00
3050	FLOX (30-70)	H2		395	4.4	0.31 E	81-00-19-00
2600	FLOX (30-70)	NH3		313	1.8	0.95 E	64-00-36-00
2900	FLOX (30-70)	N2H4		323	1.2	1.12 E	55-00-45-00
3410	FLOX (30-70)	RP1		316	3.0	1.08	75-00-25-00
4500	FLOX (70-30)	HYBAL A-5		348.	3.25	1.14	77-00-23-00
4250	FLOX (90-10)	CH4		354	4.7	1.04 E	82-00-18-00
4300	FLOX (90-10)	C2H6		346	3.8	1.08 E	79-00-21-00
4900	FLOX (90-10)	HYBAL A-5		359.	4.20	1.23	81-00-19-00

*Optimum sea level expansion, 1000 psia to 14.7 psia

TABLE 10a(CONT.)

Temp	Oxidizer	Fuel		I_S*	MR	S.G.	Rel. wts.
3900	FLOX (90-10)	MMH		356.0	2.65	1.23 E	73-00-27-00
4680	FLOX (90-10)	N2H4	UDMH	359.2	2.59	1.25	72-00-28-00
3800	FLOX (90-10)	UDMH		353.0	1.90	1.11 E	54-00-46-00
	F2	ALH3		347.6	3.00	1.50	75-00-25-00
5146	F2	BEH2		376.0	5.0	1.24 O	83-00-17-00
	F2	B2H6		371	5.60	1.105	85-00-15-00
5045	F2	B5H9		361	4.60	1.215	82-00-18-00
4203	F2	CH4		344	4.50	1.025	82-00-18-00
4131	F2	CH4	AL	343.9	4.29	1.02	83-00-17-00
	F2	CH4	ALH3	347.6	3.0	1.49	75-00-00-25
4320	F2	CH4	BE	344.4		1.05	80-00-17-03
5146	F2	CH4	BEH2	376.0	5.0	1.24 O	83-00-00-17
5564	F2	CH4	LI	378.0	2.65	1.00	73-00-00-27
4050	F2	C2H6		336.0	3.70	1.10 E	
4670	F2	HYDRAZOID-P		357.1	1.85	1.33	65-00-35-00
4150	F2	HYDYNE		336.0	2.15	1.22	68-00-32-00
3961	F2	H2		410.0	8.0	0.46	89-00-11-00
3961	F2	H2	AL	410.0	8.0	0.46	89-00-11-00
3961	F2	H2	ALH3	410.0	8.0	0.46	89-00-11-00
3961	F2	H2	BE	410.0	8.0	0.46	89-00-11-00
2067	F2	H2	BEH2	437.6	1.22	0.20 O	55-00-30-15
	F2	H2	B5H9	409.3	4.00	0.35	80-00-16-04
	F2	H2	LI	431.	1.04	0.19	51-00-30-19

*Optimum sea level expansion, 1000 psia to 14.7 psia

TABLE 10a (CONT.)

Temp	Oxidizer	Fuel		I_S*	MR	S.G.	Rel wts.
5564	F2	LI		378.0	2.65	1.00	73-00-27-00
4886	F2	LIH		363.0	4.56	1.31	82-00-18-00
4963	F2	MGH2		329.0	2.57	1.50	72-00-28-00
4392	F2	MMH		346.0	2.48	1.25	71-00-29-00
4862	F2	MMH	BEH2	363.4	3.35	1.25	77-00-16-07
4587	F2	NH3		359	3.30	1.175	77-00-23-00
4511	F2	NH3	AL	359.0	3.15	1.17	76-00-24-00
4815	F2	NH3	BE	362.5	3.35	1.20	77-00-18-05
5146	F2	NH3	BEH2	376.0	5.0	1.240	84-00-00-16
5258	F2	NH3	LI	373.0	2.57	1.00	72-00-11-17
4650	F2	N2H4		363.0	2.30	1.305	70-00-30-00
4661	F2	N2H4	AL	364.0	2.3	1.31	69-00-31-00
4661	F2	N2H4	ALH3	364.0	2.3	1.31	69-00-31-00
5150	F2	N2H4	BEH2	376.4	4.85	1.25	83-00-02-15
5564	F2	N2H4	LI	377.8	2.65	1.00	73-00-00-27
4508	F2	N2H4	UDMH	349.6	2.40	1.25	70-00-15-15
4411	F2	RP1		318.0	2.60	1.21	72-00-28-00
4292	F2	UDMH		343.0	2.50	1.19	71-00-29-00
2722	H20	BEH2		328	1.62	0.834	62-00-38-00
3394	H202	(HBEBH4)2		363	0.70	0.84	41-00-59-00
	H202	BEH2		357	1.50	0.975	60-00-40-00
2658	H202	B2H6		329	1.90	0.805	66-00-34-00
3020	H202	B5H9		307.6	2.33	1.04	70-00-30-00

*Optimum sea level expansion, 1000 psia to 14.7 psia

TABLE 10a (CONT.)

Temp	Oxidizer	Fuel		I_s^*	MR	S.G.	Rel. wts.
3020	H202	B5H9	AL	307.6	2.33	1.04	70-00-30-00
	H202	B5H9	ALH3	318.1	0.89	1.47	47-00-00-53
	H202	B5H9	BE	314.8	2.57	1.06	72-00-21-07
2750	H202	B5H9	LI	324	1.78	0.81	64-00-23-13
B	H202	CH4		281.	7.95	1.13	89-00-11-00
	H202	CH4	AL	292.9	1.17	1.17	54-00-12-34
	H202	CH4	ALH3	319.0	0.89	1.40	47-00-02-51
	H202	CH4	BE	326.0	1.44	0.83	59-00-21-20
B	H202	CH4	LI	281.	7.95	1.13	89-00-11-00
2439	H202	HYBAL A5		318.0	1.00	0.98 0	50-00-50-00
1825	H202	HYBAL B3		306.	1.20	0.93	55-00-45-00
2350	H202	HYBAL B3	BEH2	350.	1.0	0.90	50-00-25-25
2404	H202	H2		322	7.3	0.43	88-00-12-00
	H202	H2ALB3H8		327	1.70		63-00-37-00
2711	H202	LIH		262.0	4.26	1.26	81-00-19-00
3073	H202	MGH2		280.0	0.64	1.45	39-00-61-00
B	H202	NH3		270.9	2.89	1.11	74-00-26-00
	H202	NH3	AL	290.9	0.54	1.12	35-00-30-35
	H202	NH3	ALH3	318.6	0.69	1.29	41-00-12-47
	H202	NH3	BE	326.4	0.67	1.24	40-00-38-22
	H202	NH3	LI	277.5			39-00-30-31
C	H202	N2H4		282.0	2.17	1.26	68-00-32-00

*Optimum sea level expansion, 1000 psia to 14.7 psia

TABLE 10a (CONT.)

Temp	Oxidizer		Fuel		I_S*	MR	S.G.	Rel. wts.
2885	H202		N2H4	BE	317.4	0.82	1.15	45-00-46-09
2777	H202		N2H4	BEH2	327.3	0.43	0.96	30-00-56-14
2755	H202	H20	C6H14	BEH2	334.2	.93		98-02/ -
2897	H202	H20	HYDYNE		276.0	4.70	1.27	95-05/00-00
2300	H202	H20	H2		314.0	7.50	0.44	95-05/00-00
2937	H202	H20	H20	BEH2	344.9	0.3	0.88	98-02/50-50
2911	H202	H20	MMH		279.0	3.58	1.25	95-05/00-00
2765	H202	H20	MMH	BEH2	336.0	0.61		98-02/65-35
2506	H202	H20	NH3		262.0	3.00	1.12	95-05/00-00
2870	H202	H20	NH3	BEH2	352.0	0.76	0.86	98-02/50-50
2330	H202	H20	N2H4		282.	2.17	1.26	95-05/00-00
2908	H202	H20	RP1		272.0	7.26	1.30	95-05/00-00
2925	H202	H20	UDMH		278.0	4.52	1.24	95-05/00-00
2975	H202	H20	UDMH		276.8	3.5	1.22	78-00-22-00
3738	IRFNA		B5H9		292.3	3.50	1.17	78-00-22-00
2603	IRFNA		HYBAL A5		312.5	0.80	0.97	44-00-56-00
C	IRFNA		HYDYNE		269.3	3.30	1.32	77-00-23-00
C2030	IRFNA		H2		326.0	6.00	0.39	86-00-14-00
	IRFNA		JPX		266.0	4.30	1.31	81-00-19-00
3123	IRFNA		MMH		274.6	2.6	1.28	72-00-28-00
B2185	IRFNA		NH3		260	2.10	1.10	68-00-32-00
B2530	IRFNA		N2H4		283	1.45	1.28	59-00-41-00
2720	IRFNA		RP1		268	4.80	1.35	83-00-17-00

*Optimum sea level expansion, 1000 psia to 14.7 psia

TABLE 10a (CONT.)

Temp	Oxidizer	Fuel		I_S*	MR	S.G.	Rel- wts.
3151	IRFNA	UDMH		272.4	3.10	1.26	76-00-24-00
	MDFNA	B5H9		294	2.80	1.14	74-00-26-00
	MDFNA	DETA		270	3.14	1.39	76-00-24-00
3249	MDFNA	HYDYNE		275.4	2.80	1.32	74-00-26-00
	MDFNA	MMH		280	2.40	1.30	71-00-29-00
	MDFNA	UDMH		278	2.93	1.28	75-00-25-00
	MON(75-25)	B2H6		319.0	2.95	0.91 E	75-00-25-00
	MON(75-25)	B5H9		303.0	3.45	1.09 E	78-00-22-00
	MON(75-25)	CH4		286.0	5.15	1.02 E	84-00-16-00
	MON(75-25)	HYBAL A-5		302.0	2.30	1.09 E	70-00-30-00
	MON(75-25)	HYBAL A-5		306.0	0.80	0.93 E	44-00-56-00
	MON(75-25)	H2		343.8	5.85	0.37 E	85-00-15-00
	MON (75-25)	MMH		292.1	2.22	1.17	69-00-31-00
	MON(75-25)	NH3		272.3	2.10	1.04 E	68-00-32-00
	MON (75-25)	N2H4		295.1	1.4	1.19	58-00-42-00
	MON(75-25)	N2H4	UDMH	293.0	2.10	1.18 E	68-00-16-16
	MON(75-25)	RP-1		279.0	4.18	1.21 E	81-00-19-00
	MON(75-25)	UDMH		288.0	2.70	1.15 E	73-00-27-00
	MON (85-15)	B5H9		302	3.06	1.07	75-00-25-00
3893	MON (85-15)	CH2	ALH3	293.5	2.33	1.33	70-00- -
C3405	MON (85-15)	CH2	CN6H8	281.4	1.7	1.42	63-00-31-06
C3425	MON (85-15)	CH2	CN9H9	281.0	1.5	1.38	60-00-34-06
B	MON(85-15)	HYDYNE		286.0	2.90	1.20	74-00-26-00

*Optimum sea level expansion, 1000 psia to 14.7 psia

TABLE 10a (CONT.)

Temp	Oxidizer		Fuel		I_S*	MR	S.G.	Rel- wts.
	MON(85-15)		MMH		290.0	2.20	1.18	69-00-31-00
B	MON (85-15)		UDMH		288	2.64	1.16	73-00-27-00
4142	NFO2		B5H9		310.43	3.50	1.18	78-00-22-00
3343	NFO2		N2H4		295.35	1.40	1.27	58-00-42-00
4047	NFO2	N2O4	AL(BH4)2B3H8		315.4	3.0		37-37-26-00
3446	NFO2	N2O4	N2H4	UDMH	293.6	2.00	1.31	53-13-17-17
4225	NFO2	N2O4	B5H9		307.0	3.20	1.14	61-15-24-00
4956	NF3		ALH3		323.0	3.55	1.58	78-00-22-00
4731	NF3		BEH2		359.0	4.41	1.23	82-00-18-00
4809	NF3		B5H9		304.	6.7	1.37	87-00-13-00
4744	NF3		CH2	BE	307.4	4.0		80-00-04-16
4538	NF3		CH2	LI	321.0	3.5		78-00-05-17
3876	NF3		H2		351.	13.3	0.62	93-00-07-00
	NF3		LI		340.0	3.30	1.07	77-00-23-00
4384	NF3		LIH		319.0	5.90	1.36	86-00-14-00
4620	NF3		MGH2		302.0	3.55	1.52	78-00-22-00
	NF3		NH3	LI	340.0	3.30	1.07	77-00-00-23
4242	NF3		N2H4		322.	2.70	1.34	73-00-27-00
	NF3		N2H4	LI	340.0	3.30	1.07	77-00-00-23
3964	NF3		UDMH		309.	3.16	1.26	76-00-24-00
4762	N2F4		B2H6		340.2	8.0	1.28	89-00-11-00
	N2F4		B5H9		333.5	7.3	1.37	88-00-12-00

*Optimum sea level expansion, 1000 psia to 14.7 psia

TABLE 10a (CONT.)

Temp	Oxidizer			Fuel		I_S*	MR	S.G.	Rel. wts.
	N2F4			B5H9	LI	348.	3.7	1.14	78-00-00-22
3782	N2F4			CH4		314.1	6.18	1.17	86-00-14-00
	N2F4			CH4	AL	314.1	6.13	1.17	86-00-14-00
	N2F4			CH4	LI	348.	3.7	1.14	78-00-00-22
3550	N2F4			C2H6		310.0	5.02	1.24 E	83-00-17-00
3669	N2F4			HYDYNE		313.0	3.12	1.27	76-00-24-00
3842	N2F4			H2		361.0	12.00	0.59	92-00-08-00
	N2F4			LI		348.	3.7	1.14	78-00-00-22
4019	N2F4			MMH		321.0	3.25	1.28	77-00-23-00
4183	N2F4			NH3		321.0	4.00	1.23	80-00-20-00
	N2F4			NH3	AL	327.5	4.25	1.27	81-00-15-04
	N2F4			NH3	LI	348.0	3.70	1.14	78-00-00-23
4408	N2F4			N2H4		332.0	3.06	1.43	75-00-25-00
4253	N2F4			N2H4	UDMH	322.9	3.30	1.30	77-00-12-12
3950	N2F4			RP1		299.0	3.50	1.26	78-00-22-00
3986	N2F4			UDMH		316.0	3.10	1.22	76-00-24-00
4674	N2F4	CN3F7	CL03F	B5H9		323	8.2	1.39	42-42-15/11
3933	N2F4	CN3F7	CL03F	CH4		306.2	7.12	1.23	88-00-12-00
4225	N2F4	CN3F7	CL03F	HYDRAZOID-P		316.8	2.71	1.45	73-00-27-00
3599	N2F4	CN3F7	CL03F	H2		348.1	11.2	0.58	42-42-15/08
4164	N2F4	CN3F7	CL03F	MMH		312.7	3.70	1.38	78-00-21-00
4059	N2F4	CN3F7	CL03F	NH3		311.0	4.82	1.32	83-00-17-00
	N2F4	CN3F7	CL03F	N2H4		316.	3.30	1.43	42-42-15/28

*Optimum sea level expansion, 1000 psia to 14.7 psia

TABLE 10a (CONT.)

Temp	Oxidizer			Fuel		I_S*	MR	S.G.		Rel. wts.
3746	N2F4	CN3F7	CL03F	RP-1		290.1	3.99	1.36		80-00-20-00
4783	N2H4			BE		329.	3.00	1.14		75-00-25-00
2840	N2H4			B5H9		328	1.27	0.80		56-00-44-00
4550	N2H4			HYBAL A5		319.	5.00	0.95		83-00-17-00
3350	N2H4	H202		BE		335	0.54	1.24		- -65-00
B3367	N204			(CH3)2NH		279.3	3.40	0.77		77-00-23-00
B3322	N204			(CH3)20		275.0	2.63	0.82		72-00-28-00
B3082	N204			(HOCH2)2		257.0	1.80	1.21		64-00-36-00
B3360	N204			(NH2)2C2H4		280.2	2.72	1.14		73-00-27-00
4375	N204			AL		237.2	2.90	1.80	0	74-00-26-00
3835	N204			AL(BH4)2B3H8		315.2	2.4			71-00-29-00
3702	N204			ALH3		299.4	.93	1.45	0	48-00-52-00
3572	N204			AL2N2C5B6H38		310.5	1.70	1.05		63-00-37-00
4351	N204			B		258.1	3.18	1.59	0	76-00-24-00
3355	N204			BE		326.	0.52	1.69		34-00-66-00
3077	N204			BEH2		328.7	2.00	1.04		67-00-33-00
B2725	N204			B10H13C2H5		291.3	3.34	1.22		77-00-23-00
3605	N204			B2H6		316.6	2.85	.90	0	74-00-26-00
4030	N204			B5H9		299.7	3.35	1.11		77-00-23-00
4030	N204			B5H9	AL	299.7	3.35	1.11		77-00-23-00
3992	N204			B5H9	ALH3	300.	1.77	1.26		64-00-11-25
4030	N204			B5H9	BE	299.7	3.35	1.11		77-00-23-00
3890	N204			B5H9	BEH2	336.6	1.77	1.01	0	64-00-00-24

*Optimum sea level expansion, 1000 psia to 14.7 psia

TABLE 10a (CONT.)

Temp	Oxidizer	Fuel		I_s*	MR	S.G.	Rel- wts.
4030	N2O4	B5H9	LI	299.7	3.35	1.11	77-00-23-00
3460	N2O4	CH2		276.0	4.15	1.30 0	81-00-19-00
3900	N2O4	CH2	AL	280.5	2.03	1.41	67-00-17-16
3900	N2O4	CH2	ALH3	302.7	1.27	1.24	56-00-30-14
3492	N2O4	CH2	B	277.	3.55	1.37	78-00-15-07
3910	N2O4	CH2	BE	304.	2.22	1.34	69-00-19-12
3597	N2O4	CH2	LIALH4	295.	1.50	1.18	60-00-10-30
B3506	N2O4	CH3CN		273.5	2.55	0.93	72-00-28-00
B3280	N2O4	CH3NH2		277.9	3.05	0.80	75-00-25-00
B3348	N2O4	CH4		283.5	5.05	1.03	84-00-16-00
3864	N2O4	CH4	AL	291.1	1.78	1.16	64-00-14-22
3622	N2O4	CH4	ALH3	309.	1.27	1.16	56-00-10-34
3465	N2O4	CH4	BE	309.7	1.94	0.99	66-00-20-14
3578	N2O4	CH4	BEH2	337.	1.78	0.92	64-00-11-25
3160	N2O4	C2H4CL2		235.7	1.10	1.33	52-00-48-00
B4480	N2O4	C2H4O		277.5	2.13	1.01	68-00-32-00
B3365	N2O4	C2H5NH2		279.4	3.40	0.78	77-00-23-00
C	N2O4	C2H6		281.0			E
B3303	N2O4	C3H7OH		270.3	3.13	0.90	76-00-24-00
B3515	N2O4	C6H5NH2		270.3	3.13	1.10	76-00-24-00
B3659	N2O4	HCN		276.6	1.62	0.87	62-00-38-00
3070	N2O4	HYBAL A5		299.5	2.2	1.11	69-00-31-00

*Optimum sea level expansion, 1000 psia to 14.7 psia

TABLE 10a (CONT.)

Temp	Oxidizer	Fuel		I_S*	MR	S.G.		Rel- wts.
2175	N2O4	HYBAL A5		303.	0.70	0.92		41-00-59-00
2100	N2O4	HYBAL B3		308.	0.67	0.83		40-00-60-00
B3353	N2O4	HYDRAZOID-P		289.9	1.17	1.26		54-00-46-00
3392	N2O4	HYDYNE		282.0	2.68	1.22	0	73-00-27-00
2786	N2O4	H2		340.8	5.75	.37	0	85-00-15-00
3460	N2O4	JPX		279	3.50	1.32		78-00-22-00
B3120	N2O4	LI		258.4	1.06	.79	0	51-00-49-00
B2970	N2O4	LIH		261.2	2.82	1.21	0	74-00-26-00
3389	N2O4	MMH		288.0	2.19	1.21	0	69-00-31-00
3276	N2O4	MMH	BEH2	330.5	1.03	1.02		51-00-33-16
B2880	N2O4	NH3		269.3	2.00	.99	0	67-00-23-00
B3488	N2O4	NH3	AL	287.5	0.67	1.13		40-00-30-30
3670	N2O4	NH3	ALH3	311.	0.89	1.23		47-00-13-40
3226	N2O4	NH3	BE	315.9	0.82	0.97		45-00-38-17
3384	N2O4	NH3	BEH2	345.5	1.04	0.89		51-00-24-25
B2545	N2O4	NH3	LI	272.7	0.67	0.76		40-00-35-25
3259	N2O4	N2H4		292.2	1.30	1.22		57-00-43-00
3566	N2O4	N2H4	AL	302.3	0.54	1.34		35-00-42-23
3460	N2O4	N2H4	ALH3	317.3	0.61	1.27		38-00-32-30
3355	N2O4	N2H4	BE	326.3	0.51	1.21		34-00-52-14
	N2O4	N2H4	BEH2	337.4	1.85	1.03		65-00-02-33
B2653	N2O4	N2H4	LI	288.	0.51	0.96		34-00-49-17
3358	N2O4	N2H4	UDMH	288.1	1.95	1.20	0	66-00-17-17

*Optimum sea level expansion, 1000 psia to 14.7 psia

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TABLE 10a (CONT.)

Temp	Oxidizer	Fuel		I_s*	MR	S.G.	Rel- wts.
3372	N2O4	N2H4	UDMH	289.2	2.0	1.21	66-00-17-17
3450	N2O4	RP1		276.0	4.08	1.26 0	80-00-20-00
B3440	N2O4	UDMH		285.3	2.60	1.18 0	72-00-28-00
	N2O4	UDMH	BEH2	338.5	2.00	1.03	66-00-02-32
4938	OF2	AL		280.5	3.50	1.68 0	78-00-22-00
5976	OF2	B		330.6	3.35	1.65 0	77-00-23-00
4990	OF2	B10H13C2H5		353.8	3.84	1.30	79-00-21-00
4550	OF2	B2H6		365.6	3.60	0.99	78-00-22-00
4880	OF2	B5H9		355.0	4.00	1.19	80-00-20-00
4360	OF2	CH4		348.0	5.60	1.09	85-00-15-00
4448	OF2	C2H6		346.	4.90	1.15	83-00-17-00
4580	OF2	C3H8		354.4	4.20	1.16 0	81-00-19-00
4027	OF2	HYDRAZOID-P		333.3	1.36	1.31	58-00-42-00
4472	OF2	HYDYNE		349.0	2.75	1.27 0	73-00-27-00
3584	OF2	H2		401	6.00	0.385	86-00-14-00
4410	OF2	LIBH4		356.7	3.35	1.17 0	77-00-23-00
4240	OF2	MMH		343	2.50	1.260	71-00-29-00
3878	OF2	NH3		337.0	2.30	1.10 0	70-00-30-00
3990	OF2	N2H4		339	1.60	1.270	62-00-38-00
4006	OF2	N2H4	BE	341.8	1.24	1.29	56-00-38-06
4194	OF2	N2H4	UDMH	342.0	2.14	1.25	68-00-16-16
4566	OF2	RP-1		341	3.80	1.285	79-00-21-00
4468	OF2	UDMH		350.6	2.70	1.22 0	73-00-27-00

*Optimum sea level expansion, 1000 psia to 14.7 psia

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TABLE 10a (CONT.)

Temp	Oxidizer		Fuel		I_s*	MR	S.G.		Rel. wts.
4808	ONF3		B5H9		332.9	6.00	1.47		86-00-14-00
4485	ONF3		B5H9		309.7	6.00	1.47	Ø	86-00-14-00
4127	ONF3		MMH		321.3	3.00	1.47		75-00-25-00
3189	ONF3		NH3		276.2	3.00	1.24	Ø	75-00-25-00
3484	ONF3		N2H4		292.8	2.00	1.47	Ø	67-00-33-00
3346	ONF3		RP-1		269.2	4.00	1.49	Ø	80-00-20-00
	ONF3		UDMH		289.6	3.80	1.40	Ø	79-00-21-00
4481	ONF3	CLF3	B5H9		306.63	6.00	1.46		69-17-14-00
4074	ONF3	CN4F8	N2H4		314.1	3.0			- -25-00
4259	ONF3	C2N5F11	N2H4		321.1	3.5			35-43-22-00
4705	ONF3	N2F4	B5H9		328.1	6.00	1.43		69-17-14-00
4713	ONF3	N2F4	B5H9		322.5	7.00	1.38		44-44-12-00
	O2		(NH2.C2H4)2NH		298.2	1.2	1.05		55-00-45-00
	O2		(NH2.C2H4)2NH	C(NO2)4	299.8	1.6	1.17		62-00-34-04
4540	O2		ALH3		309.0	0.8	1.31		44-00-56-00
5614	O2		BE		256.1	1.78	1.32		64-00-36-00
4110	O2		BEH2		356.0	1.30	0.86		56-00-44-00
3855	O2		B2H6		342.6	2.15	.75		68-00-32-00
4400	O2		B5H9		320.0	2.4	.92		71-00-29-00
3590	O2		CH4		311.	3.35	.82		77-00-23-00
3541	O2		CH4	AL	311.	3.35	0.82		77-00-23-00
3885	O2		CH4	ALH3	324.	1.22	0.96		55-00-15-30

*Optimum sea level expansion, 1000 psia to 14.7 psia

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TABLE 10a (CONT.)

Temp	Oxidizer	Fuel		I_S*	MR	S.G.	Rel- wts.
3556	O2	CH4	BEH2	359.3	1.24	0.71	55-00-64-36
	O2	CH4	LI	311.0	3.35		77-00-23-00
4142	O2	C2H5B10H13		308.8	2.10	1.01	68-00-32-00
	O2	C2H6		306.7	3.0	0.90	75-00-25-00
	O2	DETA		298.2	1.2	1.05	55-00-45-00
	O2	DETA	C(NO2)4	299.8	1.6	1.17	62-00-34-04
3781	O2	HYBAL A5		333.0	1.40	0.93	58-00-42-00
3589	O2	HYDYNE		306.0	1.70	1.02	63-00-37-00
4670	O2	H2		391.0	4.00	0.28	80-00-20-00
	O2	H2	(BE-AL)	448.	0.87		46-00- -
	O2	H2	(BE-B5H9)	449.7	0.84		46-00- -
	O2	H2	(BE-N2H4)	446.5	0.50	.	33-00- -
	O2	H2	AL	396.0	.	.	- - -
	O2	H2	ALH3	396.5	0.63	.	39-00- -
2300	O2	H2	B	401.5	1.08	.24	52-00- -
2786	O2	H2	BE	457.	0.87	0.22	47-00-28-25
2785	O2	H2	BEH2	457.	0.85	0.23	46-00-24-30
	O2	H2	B5H9	399.6	1.0		50-00- -
2444	O2	H2	CH4-BE	429.3	.69	.23	41-00- -
2633	O2	H2	CH4-BE	439.4	.79	.24	44-00- -
1856	O2	H2	LI	404.	.6	.20	37-00- -
3309	O2	LIH		263.0	1.94	1.01	66-00-34-00
3076	O2	MGH2		270.0	0.45	1.33	31-00-69-00

*Optimum sea level expansion, 1000 psia to 14.7 psia

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TABLE 10a (CONT.)

Temp	Oxidizer	Fuel		I_S*	MR	S.G.	Rel- wts.
3581	02	MMH		312.0	1.45	1.02	59-00-41-00
3061	02	NH3		294.0	1.36	0.89	58-00-42-00
3613	02	NH3	AL	301.7	.97	.92	49-00- -
	02	NH3	ALH3	324.8	.67	1.04	40-00- -
3324	02	NH3	BE	330.6	.53	.86	35-00- -
3583	02	NH3	BEH2	360.5	.80	.79	44-00- -
	02	NH3	LI	294.0	1.35	0.89	58-00-42-00
3450	02	N2H4		313.0	0.90	1.07	47-00-53-00
3613	02	N2H4	AL	316.4	.74	1.14	42-00- -
	02	N2H4	ALH3	330.0	.52	1.17	34-00- -
3536	02	N2H4	BE	337.5	0.39	1.12	28-00-56-16
3791	02	N2H4	BEH2	361.5	.71	.92	41-00- -
3541	02	N2H4	LI	311.0	3.35	0.82	77-00- -
3523	02	N2H4	UDMH	311.5	1.29	1.02	56-00-22-22
3690	02	RP1		300.6	2.6	1.02	72-00-28-00
3614	02	UDMH		310.0	1.67	0.97	63-00-37-00
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*Optimum sea level expansion, 1000 psia to 14.7 psia

Table 10b contains a summary of performance information on the solid propellant combinations. The fuels and oxidizers of Table 5 were combined along with binder materials from Appendix A. Values of specific impulse, propellant density, and combustion temperature are listed. Table 10b actually lists families of propellants rather than individual compositions. Each family includes many formulations, but for comparative purposes, they can be grouped into classes based on one or two key ingredients, e.g. ammonium perchlorate plus aluminum or beryllium; or beryllium hydride plus any oxidizer. Also, minor variations have been omitted for the sake of brevity and clarity. For example, in addition to beryllium hydride and aluminum hydride, various other metal hydrides have been considered or even carried through to propellant formulations. Lithium aluminum hydride is one which has perhaps received most attention. However, these have been universally less desirable than the basic aluminum hydride or beryllium hydride either because of lower attainable specific impulse or incompatibility with other propellant ingredients or both.

Propellant combinations with both lower specific impulse and bulk density were immediately rejected. Where both values were higher, the propellant combination was retained. In the intermediate cases where one value is higher and the other lower, some tradeoff between the effects of specific impulse and bulk density is necessary. These tradeoffs were determined in Volume III for each stage of the nominal Apollo vehicle. Combinations of specific impulse and bulk density which result in no change in the nominal payload were determined and zero payload-change contours described. These are plotted in Fig. 2 where the specific impulse and bulk density are presented as a percent of the nominal values.

The payload gain potential for the propellant combinations was determined using Fig. 2 and the propellant performance data listed in Table 10. A propellant combination with a bulk density and specific impulse giving a point above the lines would give a payload increase and would be retained; whereas if a point below the lines occurred, the propellant combination was rejected.

When the combination of specific impulse and bulk density gives a point very near or intermediate to the lines, where mixture ratio variations could strongly affect the point, the propellant combination was retained for further study. The hydrogen-fueled combinations fell into this category and were retained in the investigation. As a result of this screening, all remaining propellant combinations will provide some increase in payload over the present N_2O_4/50-50 combinations. Later investigation will compare the actual payload increases.

TABLE 10b

SOLID PROPELLANT FAMILIES - PROPERTIES, PERFORMANCE AND AVAILABILITY

	Density lb/cu in.	T_c deg K	Specific Impulse* Theor.	Specific Impulse* Expected	Specific Impulse* Measured
Aluminum-Ammonium Perchlorate					
hydrocarbon based binder and Additives (e.g., HMX)	0.060-0.065	3000-3500	260-272		240-250
double base binder and Additives (e.g., HMX)	0.060-0.068	3100-3900	260-275		245-260
Beryllium-Ammonium Perchlorate					
hydrocarbon based binder	0.059-0.061	3100-3700	280-285		259
double base binder	0.060-	3300-	280-		
and HMX	0.061	3900	289		270
and THA	0.060	3500	285-291		269-274
Beryllium-Hydrazine Nitroform hydrocarbon based binder and AP	0.059	3400-3500	285-292		265-271
Aluminum-Hydrazine diperchlorate hydrocarbon based binder	0.066-0.070	3500-3800	270-272	250	
Nitronium perchlorate-hydrocarbon	0.062	3600	278	265	
and AL	0.064		281	264	
Aluminum Hydride-Ammonium Perchlorate					
double base binder or polynitramine binder	0.059-0.061	2300-3500	285-291	270	
hydrocarbon based binder		3200-3500	285-290	270	
Aluminum hydride-Nitronium Perchlorate double base binder	0.059	3500-3900	292-298	280	
Beryllium hydride-Ammonium Perchlorate					
double base binder or polynitramine	0.040-0.045	3100-3500	310-320	295	
hydrocarbon based binder	0.035-0.045	2800-3300	310-320	295	
Difluoramino Oxidizers					
NFPA-TMETN-AP-RDX-Al	0.065		273		259
NC-DPN-AP-Al-B			265		246
Possible $-NF_2$-B-HNF or NP	0.065	3700-4200	285-298	275-285	
Hypothetical $-NF_2$-Li-NP or Be	0.05	3600-4600	290-315	278-303	

*Chamber Pressure/Exit Pressure = 1000/14.7

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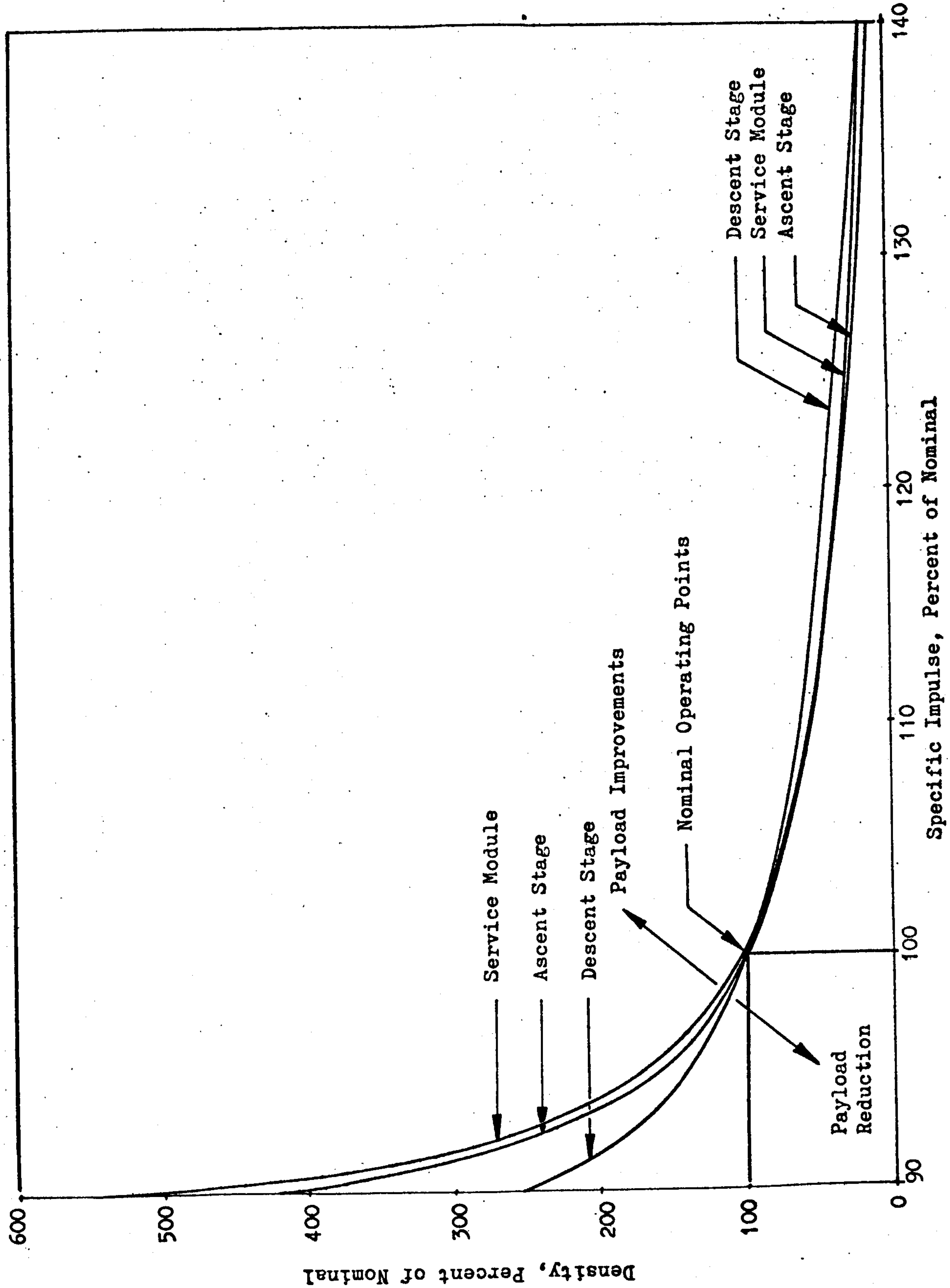

Fig. 2 Specific Impulse—Density Equivalence for Apollo Propulsion Systems

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LIQUID PROPELLANT COMBINATION COMPARISON FACTORS

From the preliminary screenings, a number of propellant combinations remain, each providing some increase in payload, and each potentially able to be developed in one of the time periods. To compare these propellant combinations further and to ensure that this comparison proceeds in a rational manner, a comparison and rating procedure was developed.

Five major areas in which comparisons could be made were established: (1) Performance, (2) Reliability, (3) Operational Aspects, (4) Development Ease, and (5) Launch Operation Ease. Each of these areas were composed of specific comparison factors. These factors represent the various propulsion system characteristics. They are listed in Table 11 . The factors in themselves combine several propellant or propulsion-system properties.

In this section, the development of the factors is presented and the evaluation of the factors for the different propellant combinations is described. The grouping of the factors into the five basic areas of comparison, and the overall rating system description is presented in the next section of this report.

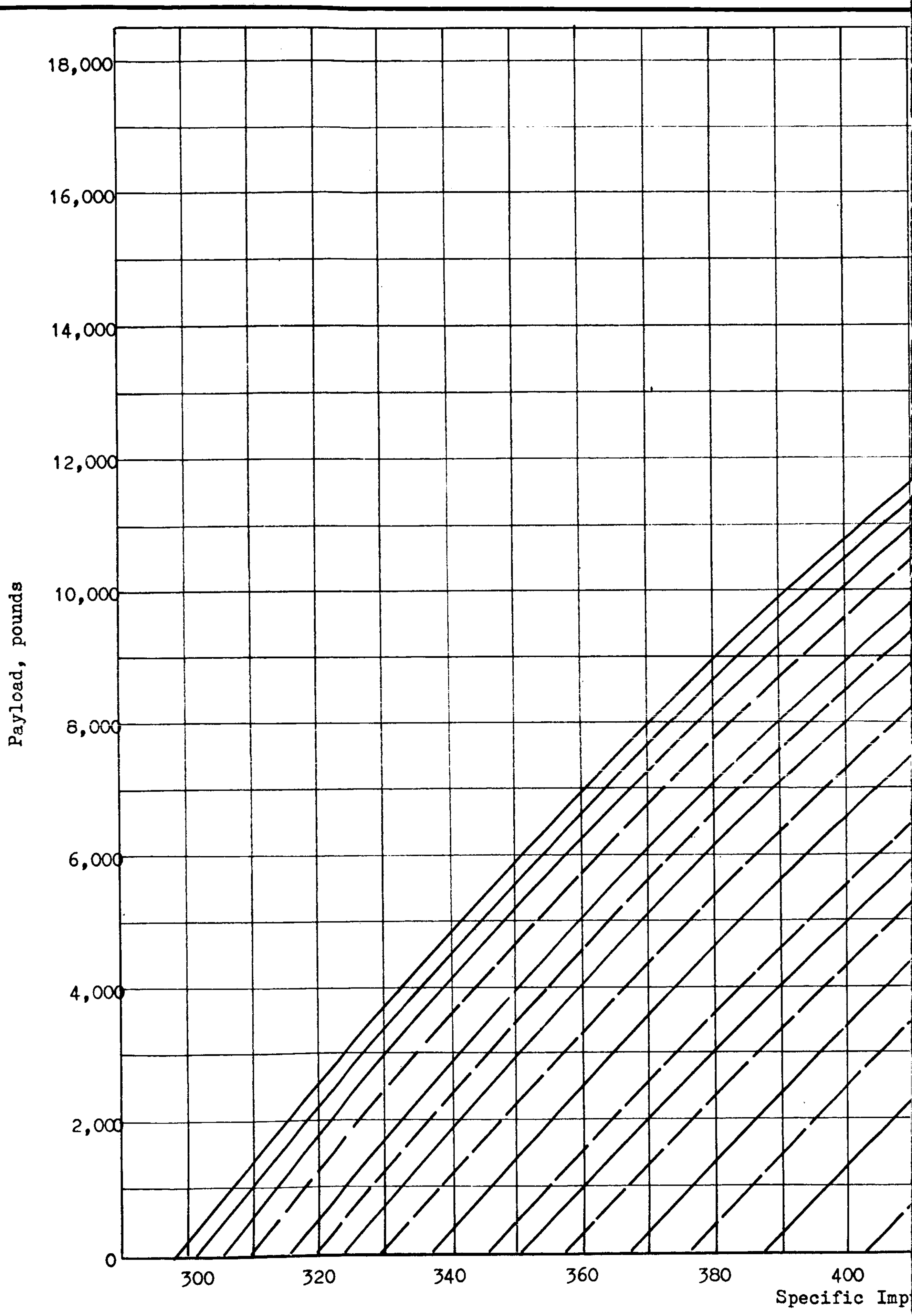
RELATIVE PAYLOAD CAPABILITY

All of the propellant combinations remaining in contention will provide some increase in payload over the present N_2O_4/50-50 combination (The present landed payload is very small $<$300 pounds.) The actual payload increase provided by a propellant combination is however, an important area of propellant combination comparison. A comparison factor was, therefore, established to provide an indication of the relative payload capability of the various propellant combinations. From the investigation of the model Apollo mission and vehicle (Volume III), the effects of specific impulse and bulk density on landed payload increase (over N_2O_4/50-50) were described assuming the same propellant (specific impulse and bulk density) used in all three Apollo propulsion systems (Fig. 3). Using this figure, payload increase estimates were made for each propellant combination.

TABLE 11

PROPELLANT COMPARISON FACTORS

1. Relative Payload Capability
2. Relative Propellant Volume
3. Propulsion System Experience
4. Propellant Physical State
5. Propulsion System Simplicity
6. Propulsion System Sensitivity
7. Propellant Thermal Storage in Space
8. Propellant Toxicity
9. Propellant Logistics
10. Thrust Chamber Cooling
11. Propellant Storage at Launch



18,000
16,000
14,000
12,000
10,000
8,000
6,000
4,000
2,000
0
Payload, pounds
300
320
340
360
380
400
Specific Imp

FORM R 18-G-18

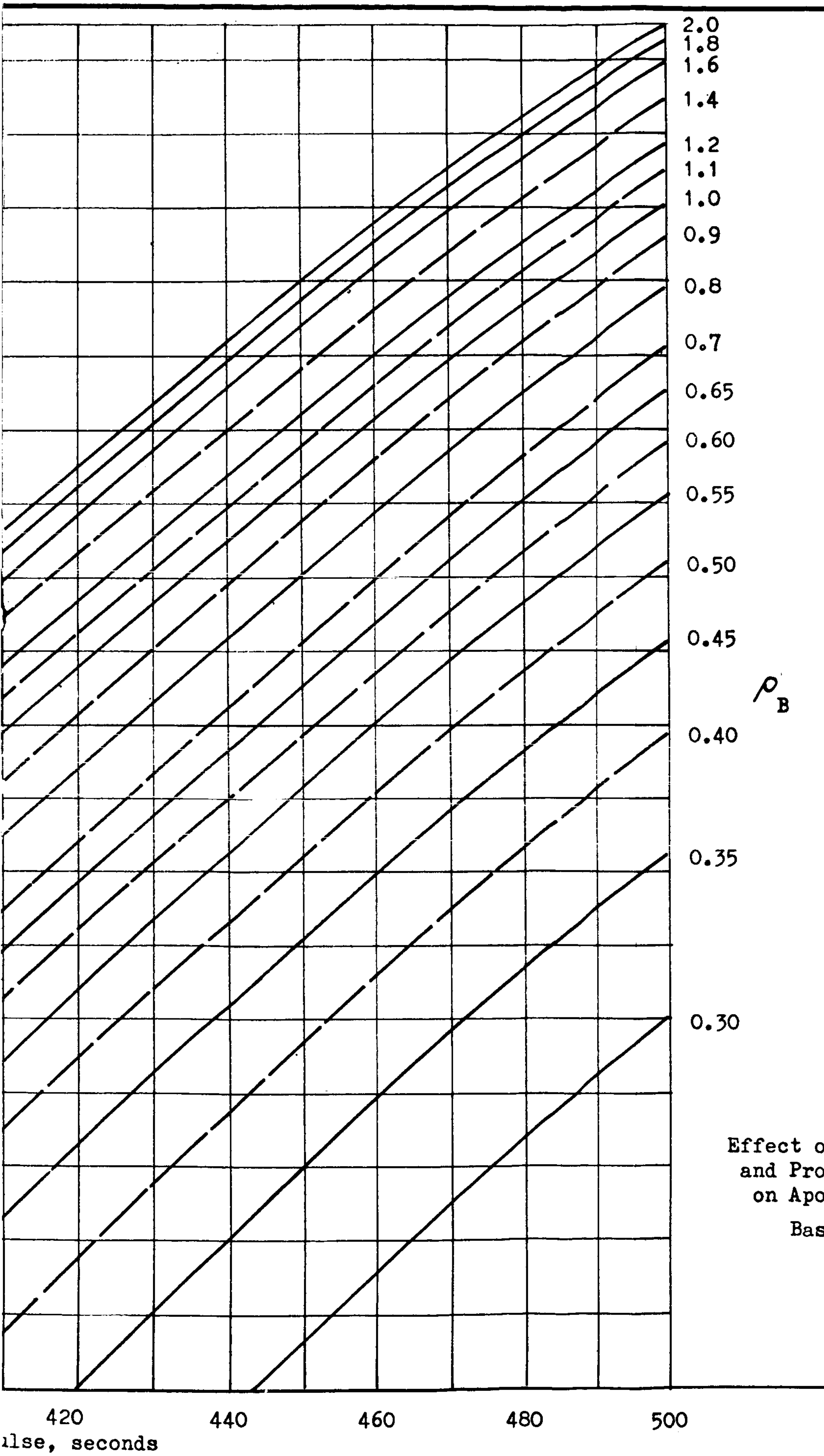


Fig. 3

Effect of Engine Specific Impulse
and Propellant Specific Gravity
on Apollo Spacecraft Payload

Based on Analysis in Volume III.

Specific impulse and bulk-density values were based upon the values listed in Table 10 (Chemical equilibrium was assumed.) In order to obtain an estimate of the specific impulse at vacuum conditions, the values of Table 10 were scaled up by a factor which related the optimum expansion specific impulse (1000 psia to 14.7 psia) to vacuum specific impulse (chamber pressure = 300 psia; expansion ratio = 40:1) for N_2O_4/50-50. An efficiency of 93 percent was placed on the specific impulse for all propellant combinations. This simplified approach is justified by the extremely large number of propellants considered and the fact that optimum expansion specific impulse values were more frequently available than the vacuum values. The payload was, therefore, an approximate value and slightly biased against the high-energy, flourine-type propellants which probably achieve higher efficiencies. The highest payload encountered was approximately 10,000 pounds. This value was assigned a rating of ten and a linear payload-rating relation used between this value and the minimum, or zero, value.

Where one of the propellants in a combination has a very low density (i.e., LH_2), the payload capability estimated by Fig. 3 is not realistic since the weight factors (based upon the current Apollo) are too high for use with low-density propellant. For these propellants, lower tank weights are necessary and the effect of mixture ratio on performance must be investigated.

The dependency of the payload capability upon the mixture-ratio and the tank factor is illustrated for four representative propellant combinations, in Figs. 4 and 5 . Mixture ratios about the optimum were used. The tank factor was represented as a percent of the nominal tank weight. Both mixture ratio and tank factor could be varied to obtain a comparative payload. However, since all other propellant combinations were compared on the basis of payload capability at the mixture ratio to maximize specific impulse, the mixture ratio of the hydrogen fueled systems was not altered for determining the comparative payload. The tank factor for all hydrogen fueled systems was reduced by 50 percent in each of the three stages. The dependency of the payload capability on mixture ratio and tank weight factor is illustrated for four representative propellant combinations in Figs. 4 and 5 . Tank factors are presented as a percent of the nominal values. The hydrogen-fueled combinations are, as expected, extremely dependent on the tank factor and mixture ratio. To account for these effects, the payload capability of the hydrogen-fueled combinations was evaluated at a tank factor representing 50 percent of the moninal value. The mixture ratio providing maximum specific impulse was used since most of the available data was for this condition.

The relative payload comparison factors are presented in Tables 12 and 13. Although these are estimates, they should reflect the relative payload capability accurately. In Table 12 the X indicates that the propellant provides less payload than N_2O_4/50-50.

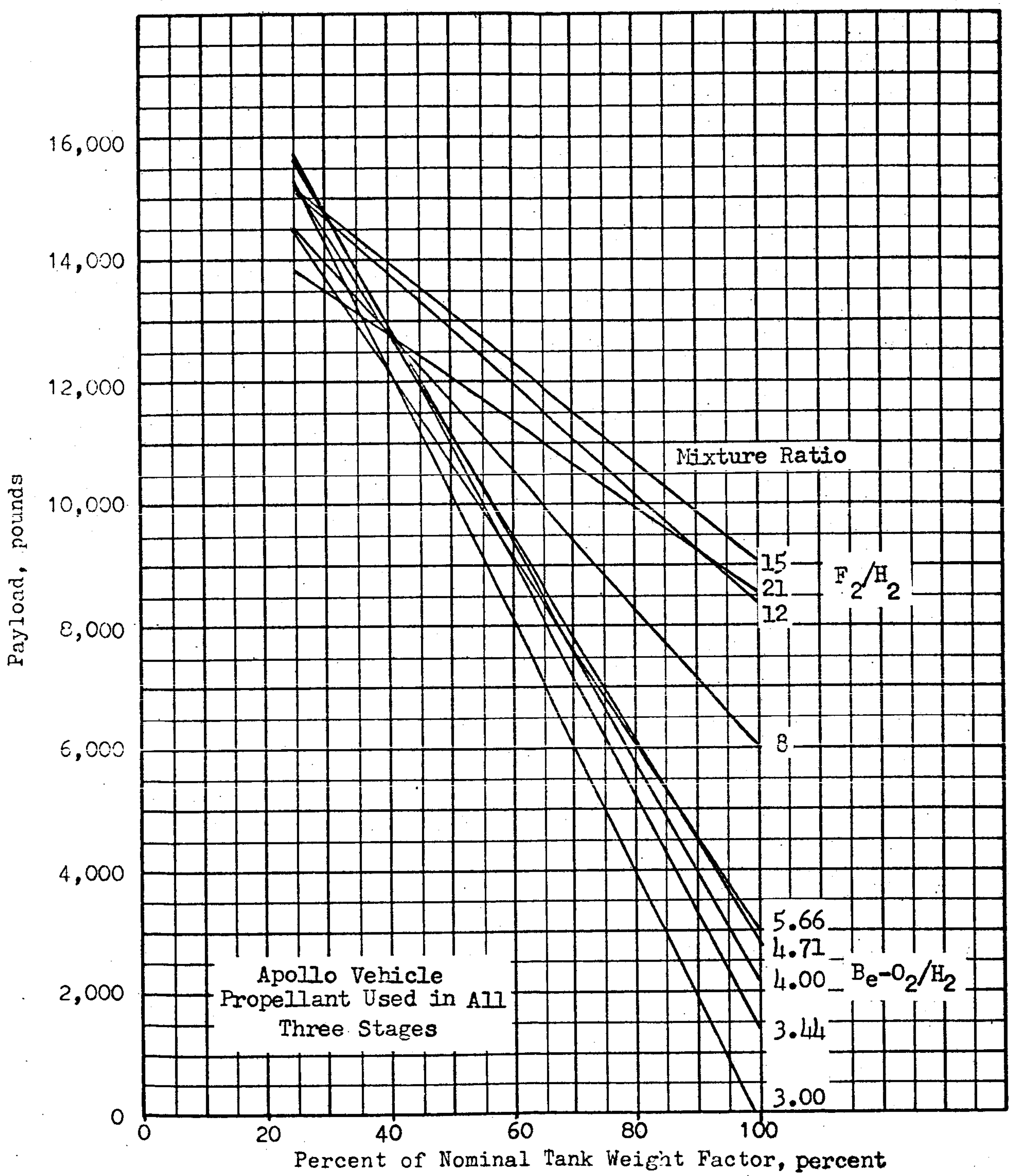


Fig. 4 Effect of Tank Weight Factor on Payload

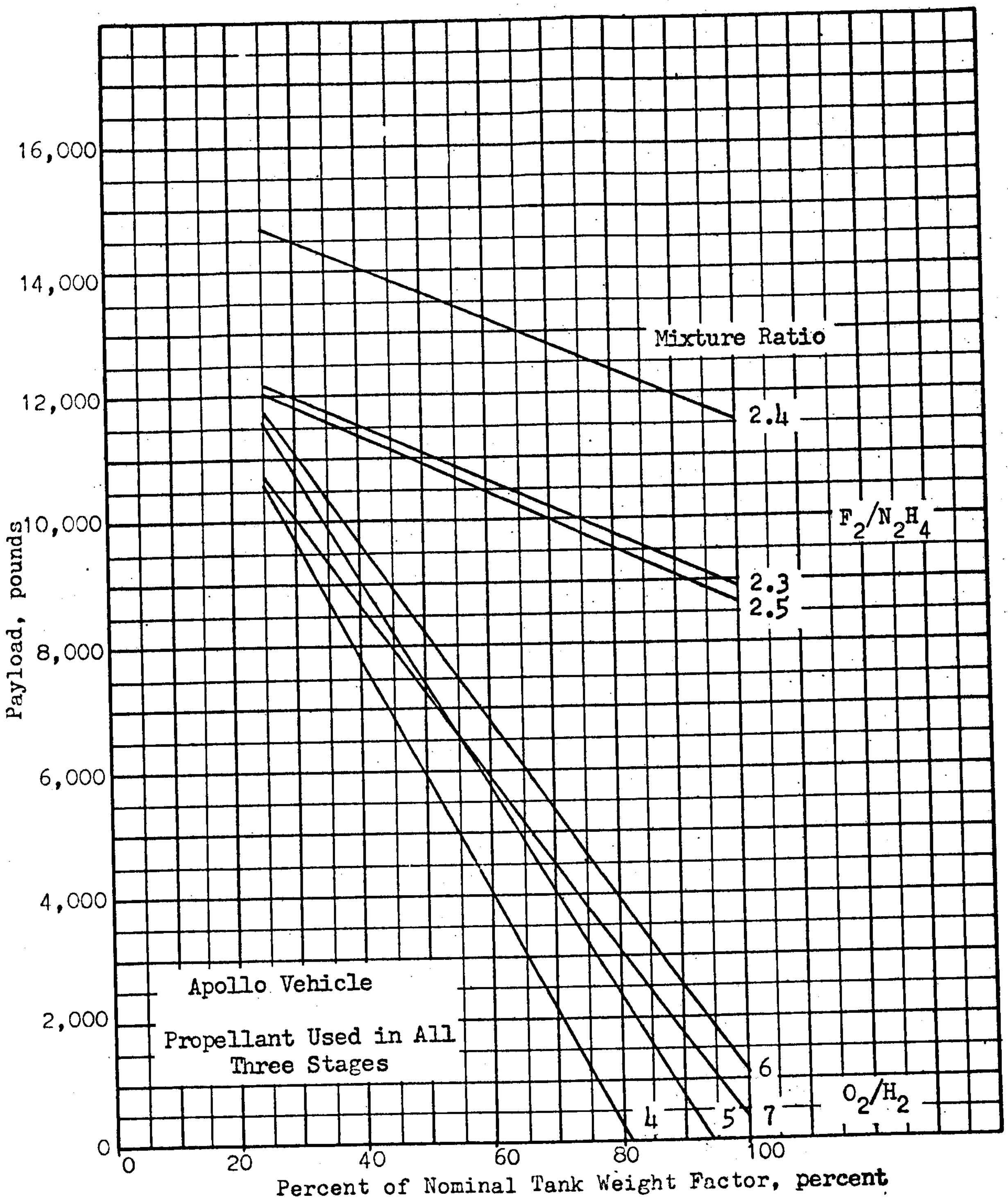


Fig. 5 Effect of Tank Weight Factor on Payload

TABLE 12

PAYLOAD AND RELATIVE VOLUMES

Oxidizer	Fuel	MR	Bulk Specific Gravity	Payload	$\frac{V_p}{V_p \text{ Ref.}}$	$\frac{V_o}{V_o \text{ Ref.}}$	$\frac{V_f}{V_f \text{ Ref.}}$	Volume Rating Factor
ClF_3	B_2H_6	7.0	1.33	1400	0.89	1.01	0.74	
	B_5H_9	7.0	1.47	870	0.82	1.04	0.54	8.2
	CH_4	5.66	1.22	X	--	--	--	--
	Hybaline A-5	5.0	1.47	990	0.82	0.98	0.60	8.5
	Hydrazoid P	2.3	1.52	1300	0.80	0.82	0.80	8.9
	Hydyne	2.93	1.42	X	--	--	--	--
	H_2	11.72	0.62	--	1.81	1.02	2.8	0
	MMH	2.7	1.42	900	0.84	0.87	0.84	8.7
	NH_3	3.65	1.34	X	0.91	0.98	0.89	8.2
	N_2H_4	2.9	1.52	1300	0.78	0.87	0.70	8.8
	N_2H_4 MMH	2.87	1.44	X	--	--	--	--
	N_2H_4 UDMH	2.9	1.45	300	0.84	0.88	0.78	8.7
	RP-1	3.26	1.41	X	--	--	--	--
	UDMH	3.1	1.38	X	0.89	0.92	0.88	8.5
ClO_3F	B_5H_9	4.0	1.17	1500	0.85	1.07	0.52	7.9
ClO_3F	N_2H_4 MMH	3.0	1.34	X	--	--	--	--
ClF_5	B_2H_6	7.0	1.36	3800	0.83	0.94	0.70	
	CH_4	3.0	1.04	X	--	--	--	--
	Hybaline A-5	5.0	1.51	3500	0.76	0.91	0.59	9.4
	Hydrazoid P	2.0	1.52	3500	0.75	0.73	0.84	9.5
	MMH	2.7	1.45	2300	0.81	0.82	0.80	
	NH_3	3.8	1.27	1700	0.91	0.91	0.81	
	N_2H_4	2.7	1.47	3800	0.77	0.80	0.68	9.4
	N_2H_4 UDMH	2.72	1.46	2500	0.80	0.81	0.78	
ClO_3F	B_2H_6	3.0	0.93	1700	1.26	1.08	1.4	8.3
	B_5H_9	4.0	1.14	3150	0.98	1.14	0.81	8.9
	Hydyne	2.8	1.22	X	--	--	--	--
	H_2	6.0	0.38	--				0
	MMH	2.24	1.20	350	0.90	0.87	0.94	
	NH_3	2.02	1.05	X	--	--	--	--

TABLE 12 (Continued)

PAYLOAD AND RELATIVE VOLUMES

Oxidizer		Fuel	MR	Bulk Specific Gravity	Payload	V_p / V_p Ref.	V_o / V_o Ref.	V_f / V_f Ref.	Volume Rating Factor
ClO_3F		N_2H_4	1.4	1.22	1700	0.96	0.86	1.06	
		N_2H_4MMH	2.2	1.2	350	0.90	0.87	0.94	
		RP-1	4.35	1.25	X	--	--	--	--
		UDMH	2.7	1.17	X	--	--	--	--
FLOX	30-70	B_2H_6	2.9	0.85	5600	1.22	1.12	1.33	4.4
	30-70	B_5H_9	3.2	1.00	4800	1.08	1.20	0.91	6.1
	30-70	H_2	4.4	0.31	--	3.3	1.13	6.0	0
	30-70	NH_3	1.8	0.95	1600	1.20	1.07	1.33	4.6
	30-70	N_2H_4	1.2	1.12	3400	0.99	0.88	1.12	7.2
	30-70	RP-1	3.0	1.08	2800	1.05	1.24	0.78	6.0
	90-10	CH_4	4.7	1.04	6800	1.00	0.95	1.04	7.8
	90-10	C_2H_6	3.8	1.08	6000	0.99	0.90	1.03	7.8
	90-10	MMH	2.65	1.23	8000	0.83	0.91	0.72	9.1
	90-10	N_2H_4 UDMH	2.59	1.25	8300	0.82	0.91	0.70	8.6
	90-10	UDMH	1.9	1.11	7500	0.88	0.91	0.80	8.5
F_2		B_2H_6	5.6	1.105	8900	0.91	1.00	0.76	8.2
		B_5H_9	4.6	1.215	8400	0.84	0.99	0.65	8.4
		CH_4	4.5	1.205	5700	1.04	1.04	1.00	7.4
		C_2H_6	3.7	1.1	5300	0.98	1.01	0.95	8.0
		Hydrazoid P	1.85	1.33	8400	0.78	1.09	0.75	8.0
		Hydyne	2.15	1.22	5700	0.88	0.88	0.90	8.5
		H_2	8.0	0.46	--	2.01	0.97	3.3	0
		MMH	2.48	1.25	7000	0.85	0.89	0.77	8.6
		NH_3	3.3	1.175	8000	0.89	0.90	0.80	8.5
		N_2H_4	2.3	1.305	8800	0.78	0.82	0.68	9.0
		N_2H_4 UDMH	2.4	1.25	7400	0.85	0.86	0.77	8.8

TABLE 12 (Continued)

PAYLOAD AND RELATIVE VOLUMES

Oxidizer	Fuel	MR	Bulk Specific Gravity	Payload	$\frac{V_p}{V_p \text{ Ref.}}$	$\frac{V_o}{V_o \text{ Ref.}}$	$\frac{V_f}{V_f \text{ Ref.}}$	Volume Rating Factor
	RP-1	2.6	1.21	3700	0.93	0.97	0.85	8.2
	UDMH	2.5	1.19	6450	0.89	0.90	0.87	8.5
H_2O_2	B_2H_6	1.9	0.805	2550	1.36	0.88	1.88	1.7
	B_5H_9	2.33	1.04	1860	1.11	1.00	1.20	6.0
	CH_4	7.95	1.13	X	—	—	—	—
	Hybaline A5	1.0	0.98	2600	1.19	0.70	1.66	3.6
	H_2	7.3	0.43	—				0
	MMH	3.58	1.25	X	—	—	—	—
	NH_3	2.89	1.11	X	—	—	—	—
	N_2H_4	2.17	1.26	X	—	—	—	—
	RP-1	7.26	1.3	X	—	—	—	—
	UDMH	4.52	1.24	X	—	—	—	—
IRFNA	B_5H_9	3.5	1.17	300	1.02	1.08	0.96	7.4
	Hybaline A5	0.8	0.97	1650	1.18	0.6	2.0	2.8
	Hydyne	3.3	1.32	X	—	—	—	—
	H_2	6.0	0.39	—				0
	JPX	4.3	1.31	X	—	—	—	—
	MMH	2.6	1.28	X	—	—	—	—
	NH_3	2.1	1.10	X	—	—	—	—
	N_2H_4	1.45	1.28	X	—	—	—	—
	RP-1	4.8	1.35	X	—	—	—	—
	UDMH	3.1	1.26	X	—	—	—	—

TABLE 12 (Continued)

PAYLOAD AND RELATIVE VOLUMES

Oxidizer	Fuel	MR	Bulk Specific Gravity	Payload	$\frac{V_p}{V_p \text{ Ref.}}$	$\frac{V_o}{V_o \text{ Ref.}}$	$\frac{V_f}{V_f \text{ Ref.}}$	Volume Rating Factor
MDFNA	B_5H_9	2.8	1.14	300	1.04	1.02	1.09	
	DFTA	3.14	1.39	X	--	--	--	--
	Hydyne	2.8	1.32	X	--	--	--	--
	MMH	2.4	1.3	X	--	--	--	--
	UDMH	2.93	1.28	X	--	--	--	--
MON (75-25)	B_2H_6	2.95	0.91	2100	1.25	1.10	1.42	
	B_5H_9	3.45	1.09	1000	1.07	1.17	0.95	6.4
	CH_4	5.15	1.02	X	--	--	--	--
	Hybaline A5	2.3	1.09	1200	1.08	1.07	1.08	
	H_2	5.85	0.37	--				0
	MMH	2.22	1.17	100	1.03	1.08	0.96	7.1
	NH_3	2.1	1.04	X	--	--	--	--
	N_2H_4	1.4	1.19	700	1.00	0.91	1.11	7.3
	N_2H_4 UDMH	2.1	1.18	500	1.01	1.04	0.97	
	RP-1	4.18	1.21	X	--	--	--	--
	UDMH	2.70	1.15	X	--	--	--	--
MOXIE 2A	B_5H_9	8.2	1.39	2300	0.81	1.19	0.44	7.9
	CH_4	7.12	1.23	2400	0.94	1.11	0.72	7.4
	Hydrazoid P	2.71	1.45	4400	0.78	0.90	0.62	8.8
	H_2	11.2	0.58	--				0
	MMH	3.7	1.38	3500	0.83	0.98	0.62	8.5
	NH_3	4.82	1.32	3200	0.86	1.04	0.66	8.1
	N_2H_4	3.3	1.43	2700	0.81	0.99	0.62	8.5
	RP-1	3.99	1.86	600	0.89	0.99	0.68	8.3

TABLE 12 (Continued)

PAYLOAD AND RELATIVE VOLUMES

Oxidizer		Fuel	MR	Bulk Specific Gravity	Payload	V_p / V_p Ref.	V_o / V_o Ref.	V_f / V_f Ref.	Volume Rating Factor
NFO_2		B_5H_9	3.5	1.18	2650	0.98	1.03	0.92	7.9
		N_2H_4	1.4	1.27	1000	0.93	0.80	1.11	7.5
	N_2O_4	N_2H_4 UDMH	2.0	1.31	1050	0.91	0.90	0.98	8.2
	N_2O_4	B_5H_9	3.2	1.14	1500	1.01	1.03	0.98	7.8
NF_3		B_5H_9	6.7	1.37	2400	0.89	1.19	0.54	7.1
		H_2	13.3	0.62	--	1.68	1.14	2.50	0
		N_2H_4	2.7	1.34	4300	0.83	0.96	0.66	8.5
		UDMH	3.16	1.26	2800	0.91	1.02	0.79	8.1
N_2F_4		B_2H_6	8.0	1.28	6400	0.99	1.26	0.76	6.2
		B_5H_9	7.3	1.37	6000	0.79	1.14	0.46	7.6
		CH_4	6.18	1.17	3000	0.97	1.18	0.83	6.9
		C_2H_6	5.02	1.24	2700	1.00	1.15	0.78	7.0
		Hydyne	3.12	1.27	3300	0.90	1.04	0.76	8.0
		H_2	12.0	0.59	--	1.73	1.15	2.70	0
		MMH	3.25	1.28	4200	0.88	1.04	0.68	
		NH_3	4.00	1.23	4100	0.91	1.08	0.73	7.7
		N_2H_4	3.06	1.43	6000	0.75	0.99	0.59	8.6
		N_2H_4 UDMH	3.3	1.3	4400	0.86	1.02	0.65	
		RP1	3.5	1.26	4100	0.93	1.10	0.72	7.5
		UDMH	3.1	1.22	3500	0.92	1.04	0.78	8.1
N_2H_4		B_5H_9	1.27	0.80	2300	1.37	1.10	1.74	2.1
		Hybaline A5	5.0	0.95	4600	1.18	1.66	0.56	3.6

TABLE 12 (Continued)

PAYLOAD AND RELATIVE VOLUMES

Oxidizer	Fuel	MR	Bulk Specific Gravity	Payload	$\frac{V_p}{V_p \text{ Ref.}}$	$\frac{V_o}{V_o \text{ Ref.}}$	$\frac{V_f}{V_f \text{ Ref.}}$	Volume Rating Factor
N_2O_4	B_2H_6	2.85	0.90	1800	1.27	1.04	1.46	3.7
	B_5H_9	3.35	1.11	990	1.06	1.12	0.99	6.8
	CH_4	5.05	1.03	X	--	--	--	--
	Hybaline A5	2.2	1.11	2100	1.06	0.94	1.14	6.7
	Hydrazoid P	1.17	1.26	200	0.96	0.85	1.19	
	H_2	5.75	0.37	--				0
	MMH	2.19	1.21	X	--	--	--	--
	NH_3	2.00	0.99	X	--	--	--	--
	N_2H_4	1.3	1.22	490	0.98	0.84	0.98	8.2
	N_2H_4UDMH	2.0	1.21	Ref	1.00	1.00	1.00	8.0
	RP1	4.08	1.26	X	--	--	--	--
	UDMH	2.6	1.18	X	--	--	--	--
OF_2	B_2H_6	3.6	0.99	7700	1.02	0.94	1.11	7.2
	B_5H_9	4.0	1.19	7610	0.87	0.97	0.74	8.4
	CH_4	5.6	1.09	6600	0.97	1.04	0.85	7.8
	C_2H_6	4.9	1.15	5500	0.92	1.02	0.74	8.0
	Hydrazoid P	1.36	1.31	5800	0.83	0.73	0.96	8.5
	Hydyne	2.75	1.27	6000	0.83	0.90	0.72	8.7
	H_2	6.0	0.385	--	2.4	0.96	4.30	0
	MMH	2.5	1.26	6650	0.85	0.88	0.77	8.7
	NH_3	2.3	1.1	4600	0.87	0.88	0.84	8.6
	N_2H_4	1.6	1.27	6250	0.85	0.78	0.92	8.6
	N_2H_4 UDMH	2.14	1.25	6600	0.86	0.85	0.86	8.7
	RP-1	3.8	1.285	6600	0.83	0.99	0.62	8.4
	UDMH	2.7	1.22	6200	0.86	0.90	0.80	8.4

TABLE 12 (Continued)

PAYLOAD AND RELATIVE VOLUMES

Oxidizer	Fuel	MR	Bulk Specific Gravity	Payload	$V_p / V_{p\ Ref.}$	$V_o / V_{o\ Ref.}$	$V_f / V_{f\ Ref.}$	Volume Rating Factor
ONF_3	B_5H_9	6.0	1.47	6300	0.74	0.88	0.56	8.9
	MMH	3.0	1.47	4800	0.76	0.80	0.71	9.1
	NH_3	3.0	1.24	--	--	--	--	--
	N_2H_4	2.0	1.47	--	--	--	--	--
	RP-1	4.0	1.49	--	--	--	--	--
	UDMH	3.8	1.4	--	--	--	--	--
O_2	B_2H_6	2.15	0.75	3500	1.41	1.22	1.70	8.0
	B_5H_9	2.4	0.92	2200	1.22	1.23	1.18	8.4
	CH_4	3.35	0.82	450	1.39	1.39	1.40	8.0
	C_2H_6	3.0	0.9	X	--	--	--	--
	DETA	1.2	1.05	500	1.13	1.00	1.26	
	Hybal A5	1.4	0.93	3800	1.18	1.00	1.40	8.5
	Hydyne	1.7	1.02	1400	1.13	1.16	1.12	
	H_2	4.0	0.28	--				0
	MMH	1.45	1.02	2100	1.12	1.07	1.20	8.6
	NH_3	1.36	0.87	X	--	--	--	--
	N_2H_4	0.9	1.07	2300	1.08	0.85	1.35	8.7
	N_2H_4 UDMH	1.29	1.02	1900	1.12	1.01	1.24	8.6
	RP-1	2.6	1.02	700	1.15	1.33	0.90	8.6
	UDMH	1.67	0.97	1400	1.18	1.12	1.27	8.5

TABLE 13

ESTIMATED PAYLOAD CAPABILITIES OF 1975 COMBINATIONS

Propellant Combination	Mixture Ratio	Payload	Propellant Combination	Mixture Ratio	Payload
F_2/H_2	12.0	11.6	ClO_3F/BeH_2	2.13	5.8
OF_2/H_2	6.0	9.4	NF_3/BeH_2	4.41	8.2
O_2/H_2	4.0	5.8	NF_3/Li	3.30	5.5
N_2F_4/H_2	12.0	8.2	$F_2/H_2/BeH_2$	1.22	6.5
F_2/AlH_3	3.0	8.1	$F_2/MMH/BeH_2$	3.35	8.8
F_2/BeH_2	5.0	9.7	$F_2/CH_4/Be$	4.0	5.8
F_2/Li	2.65	9.1	$F_2/NH_3/Be$	3.35	8.4
F_2/MgH_2	2.57	5.8	$F_2/NH_3/Li$	2.57	8.6
F_2/LiH	4.56	8.9	$F_2/N_2H_4/BeH_2$	4.85	10.0
$OF/LiBH_4$	3.35	7.0	$F_2/H_2/Li$	1.04	5.8
N_2F_4/Li	3.7	6.7	H_2O_2/Hybaline B-3/ BeH_2	1.0	5.5
N_2H_4/Be	3.0	6.0	$OF_2/N_2H_4/Be$	1.24	6.7
O_2/BeH_2	1.30	5.6	$O_2/NH_2/BeH_2$	0.80	5.7
$H_2O_2/(HBeBH_4)_2$	0.70	6.5	$O_2/N_2H_4/Be$	0.39	5.5
H_2O_2/BeH_2	1.50	6.7	$O_2/N_2H_4/BeH_2$	0.71	6.9
N_2O_4/Be	1.69	5.9	$O_2/H_2/Be$	0.87	9.8
			$O_2/H_2/BeH_2$	0.85	9.9

RELATIVE PROPELLANT VOLUME

The propellant volume is important from two aspects. First, the general size or compactness of the entire propulsion system; and, second, since the propellant is essentially being substituted into an existing vehicle, any increases in propellant volume means some system redesign. A propellant volume comparison factor was developed based upon the propellant volume relative to the volume of the existing propellant combination.

Relative propellant volumes were determined based upon an equilvalent Apollo propulsion system maneuver. The three Apollo propulsion systems must perform the maneuvers listed in Table 14 . Propellant requirements for the Service Module may be estimated by combining the two propulsion phases into an effective velocity requirement. This is described in Appendix B .

TABLE 14

APOLLO PROPULSION SYSTEM MISSIONS

Propulsion System	Service Module		LEM Decent	LEM Ascent
	ΔV_1	ΔV_2		
Mission Velocity Increment, ft/sec	4460	4300	7750	6880
Space Mission Type	Constant Gross Wt. (90,000 pounds)	Constant Payload (15,000 pounds)	Constant Gross Wt.	Constant Payload

Relative tank volumes are determined by the following equations based upon a constant gross weight vehicle:

$$W_P = W_g \left[1-\exp(-\Delta V/gI_s)\right] = W_g \left[f(I_s)\right] \quad (1)$$

$$\frac{V_P}{V_{P-ref}} = \frac{W_{P}/\rho B}{(W_P/\rho B)_{ref}} = \frac{[f(I_s)]\rho_{B-ref}}{[f(I_s)]_{ref}\,\rho_b} \quad (2)$$

$$V_F = \left(\frac{1}{MR + 1}\right) \frac{W_P}{\rho_f} \tag{3}$$

$$V_o = \left(\frac{MR}{MR + 1}\right) \frac{W_P}{\rho_o} \tag{4}$$

$$\frac{V_F}{V_{R-ref}} = \left(\frac{MR_{ref} + 1}{MR + 1}\right) \frac{\rho_{f-ref}}{\rho_f} \left[\frac{f(I_s)}{f(I_s)}\right]_{ref} \tag{5}$$

$$\frac{V_o}{V_{o-ref}} = \left(\frac{MR}{MR + 1}\right) \left(\frac{MR + 1}{MR}\right)_{ref} \frac{\rho_{o-ref}}{\rho_o} \left[\frac{f(I_s)}{f(I_s)}\right]_{ref} \tag{6}$$

As indicated in Table 14 , each propulsion system has a slightly different velocity increment. A velocity increment of 7000 fps is selected as a basis of evaluation. In Appendix C , it is shown that the range of ΔV values in Table 14 (from 7000 fps) created less than 2.0-percent variation from the value of relative volume calculated with the selected 7000 fps. The constant gross weight assumption is not precisely accurate but is sufficient for the comparison.

The volume-rating factors, evaluated for each component propellant of a combination, are presented in Table 12 along with the relative volumes of the propellant combination, the oxidizer, and the fuel. The ratings were established based upon a propellant tank redesign study of the Apollo system (Appendix D) and a consideration of the volumes resulting from the various propellant combinations. The volume-rating factor was obtained by combining the propellant relative volume and the oxidizer or fuel relative volumes according to the relationship of Fig. 6 .

Figure 6 was established based on a study of Apollo vehicle propellant volume limits (Appendix B). For 1970, the lowest propellant volume received the highest rating. A relative volume of 1.0 (equal to present volume) degraded the volume-rating factor by the value of 1.0. Additional degradation occurred as volume increased, depending upon the amount of redesign necessary. If an individual propellant relative volume was greater than 1.9, the propellant combination was eliminated from 1970 consideration since based on Appendix D extensive structural redesign would be required. The hydrogen-fueled propellant combinations were the only ones to exceed the propulsion redesign volume limit. These propellants were considered in the 1975 category. The 1975 category has no structural redesign restrictions.

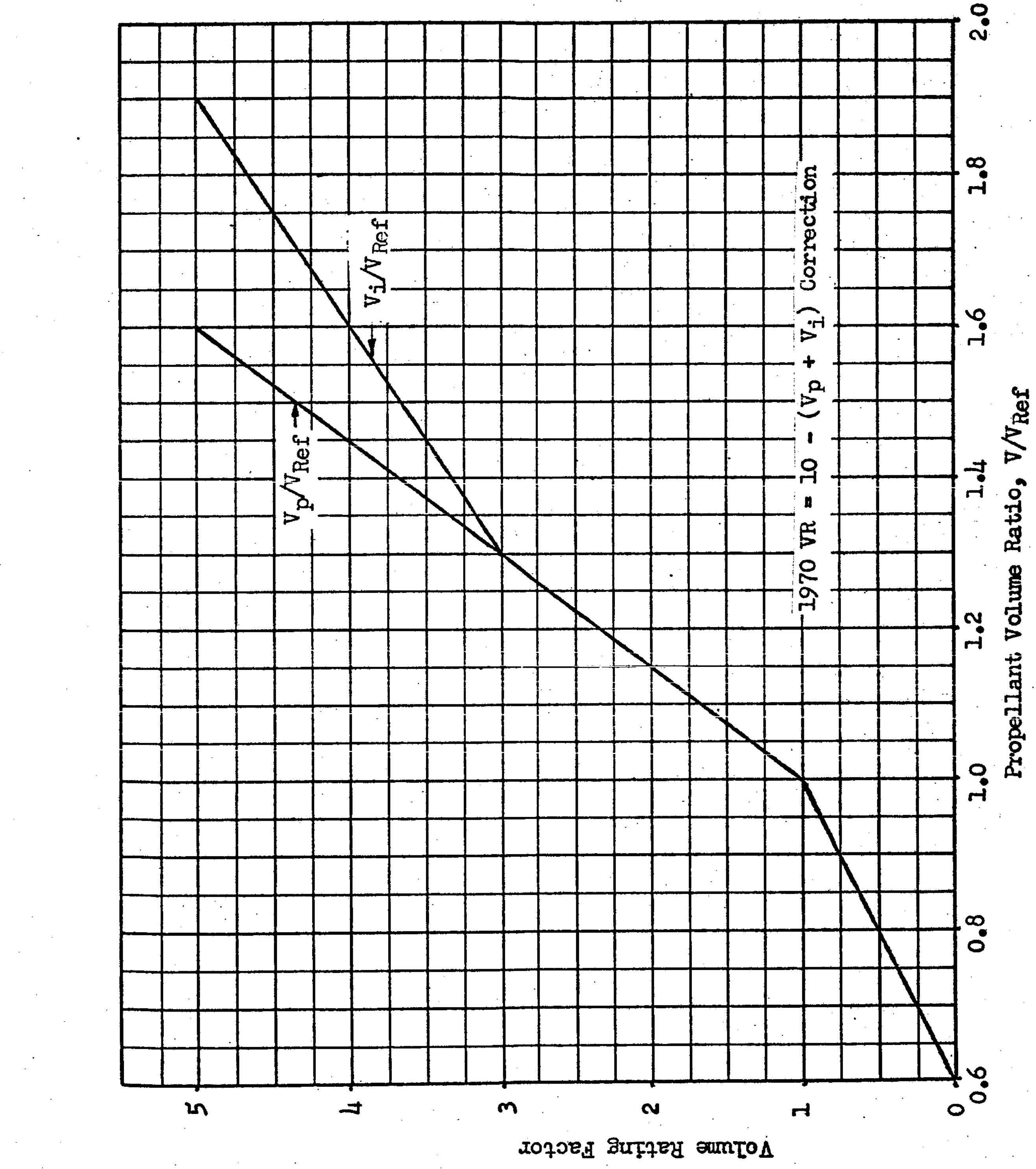


Fig. 6 1970 Volume Rating Graph

Therefore, the propellant volume was not considered in the propellant rating. Using the method described previously, volume ratings were determined. These ratings are presented in Table 12 .

PROPULSION-SYSTEM EXPERIENCE

The previous test and development experience with a propellant combination serves to indicate the state of development of technology associated with the propellant combination. Five categories of test and development experience were defined and served as a basis for the establishment of factors used to compare the relative experience that has been acquired with the different propellant combinations. These five factors and the manner in which they are "weighted" are listed in Table 15 .

TABLE 15

PROPELLANT COMBINATION TEST EXPERIENCE

Type of Testing	Total Rating
1. Propellant Property Determination	0
2. Ignition Testing	1
3. Research Thrust Chamber Tests	1
4. Component Development	
A. Thrust Chamber	2
B. Feed System: Subrating	4
1. Oxidizer 0.5, 1.0, 1.5, 2.0	
2. Fuel 0.5, 1.0, 1.5, 2.0	
5. Engine System Development	2

All of the propellants have had their basic physical properties determined. This was, therefore, selected as the lowest level of experience. The next step in developing the technology associated with a propellant combination was ignition testing. Following this level of testing, the next was small research thrust chamber testing to obtain information concerning performance, heat transfer, injector design, etc.

Further experience with a propellant combination will be either from a component or engine development program. The component development category was separated into three areas: thrust chamber, oxidizer-feed system, and fuel-feed system. The oxidizer- and fuel-feed systems were considered separately since considerable experience may have been gained in development programs using only one of the propellants in the combination under consideration. Various levels of feed-system development were considered. The ultimate in experience was assumed to occur when a complete engine-system development program has been conducted with the propellant combination under consideration.

Liquid rocket-engine systems that have been developed are listed in Table 16 along with their propellants. Looking at the experience-rating factor (Table 15), it can be seen that a rating summation of ten occurs for propellant combinations for which engine systems have been developed. At the other end of the scale (0) are the propellants for which small research thrust chamber testing and component development has occurred. The propellant-combination test experience was summarized and the experience-comparison factor evaluated. This information is presented in Table 17.

PROPELLANT PHYSICAL STATE

This rating factor refers primarily to the physical state in which the propellant is utilized and any difficulties in transferring propellants (i.e., from the storage tank to the combustion chamber) because of this property. Liquid propellant-transfer methods are well developed and solid propellants require no transfer of propellant. However, for propellants which are used in the form of gels, slurrys, or powders, there is a limited history in the development of methods for transporting the propellant to a combustion zone. Because of this, additional development effort would be necessary and the additional transfer device (if one were necessary) would inherently decrease reliability.

TABLE 16

PROPELLANTS FOR WHICH LIQUID ROCKET ENGINES HAVE BEEN DEVELOPED

Propellant	Engine	Use	Company
LO_2/RP-1	LR87AJI	Titan	AJ
	LR91AJI	Titan	AJ
	Vanguard	Vanguard Booster	GE
	G-38	Navaho	NAA
	LR79-NA9	Thor	NAA
	LR89-NA7	Atlas	NAA
	LR105-NA5	Atlas	NAA
	H-I	Saturn	NAA
N_2O_4/50-50	LR87AJ5	Titan II	AJ
	LR91AJ5	Titan II	AJ
IRFNA/UDMH	AJ10-104	Able Star	AJ
	8048	Agena A	Bell
WIFNA/UDMH	AJ10-11	Delta	AJ
IRFNA/JP-4	8001	Agena A	Bell
MON/UDMH	8101	Agena D	Bell
LO_2/LH_2	RL10A1	Centaur	P-W
H_2O_2/JP-5	TD-204	Aircraft	RMD
	A	Aircraft	NAA
LO_2/NH_3	LR-59	X-15	RMD
LO_2/Alcohol	R5-110A7	Redstone	NAA

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TABLE 17

PROPELLANT COMBINATION EXPERIENCE COMPARISON

Propellants	Ignition Tests	Research Thrust Chamber	Thrust Chamber Development	System Development Fuel	System Development Oxidizer	Engine System Development	Experience Rating
MOXIE 2A/NH_3				2.0			2.0
N_2H_4				1.5			1.5
MMH				1.5			1.5
50-50				2.0			2.0
RP-1				2.0			2.0
C_2H_6							
CH_4				1.5			1.5
B_2H_6				1.0			1.0
B_5H_9				1.5			1.5
MON /NH_3	x	x		2.0	1.0		5.0
N_2H_4	x	x	x	1.5	1.0		6.5
MMH	x	x	x	1.5	1.0		6.5
50-50	x			2.0	1.0		4.0
RP-1				2.0	1.0		3.0
C_2H_6					1.0		1.0
CH_4				1.5	1.0		2.5
B_2H_6				1.0	1.0		2.0
B_5H_9				1.5	1.0		2.5
A-5				0.5	1.0		1.5
H_2				2.0	1.0		3.0

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TABLE 17

(Continued)

Propellants		Ignition Tests	Research Thrust Chamber	Thrust Chamber Development	System Development Fuel	System Development Oxidizer	Engine System Development	Experience Rating
F_2 /	NH_3	x	x	x	2.0	1.0		7.0
	N_2H_4	x	x	x	1.5	1.0		6.5
	MMH	x	x	x	1.5	1.0		6.5
	50-50				2.0	1.0		3.0
	RP-1				2.0	1.0		3.0
	C_2H_6					1.0		1.0
	CH_4				1.0	1.0		2.0
	B_2H_6	x	x	x	1.0	1.0		6.0
	B_5H_9				1.5	1.0		2.5
	A_5				0.5	1.0		1.5
	H_2	x	x	x	2.0	1.0		7.0
	HDZP					1.0		1.0
$N_2O_2(2\%H_2O)$/	NH_3				2.0	2.0		4.0
	N_2H_4	x	x		1.5	2.0		5.0
	MMH	x			1.5	2.0		4.5
	50-50				1.5	2.0		3.5
	RP-1	x	x	x	2.0	2.0	x	10.0
	C_2H_6					2.0		2.0
	CH_4				1.0	2.0		3.0
	B_2H_6				1.0	2.0		3.0
	B_5H_9	x	x	x	1.5	2.0		7.5
	A_5				0.5	2.0		2.5
	H_2				1.0	2.0		3.0
	HDZP					2.0		2.0

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TABLE 17

(Continued)

Propellants	Ignition Tests	Research Thrust Chamber	Thrust Chamber Development	System Development Fuel	System Development Oxidizer	Engine System Development	Experience Rating
N_2H_4/B_5H_9	x	x	x	1.5	1.5		7.0
N_2F_4/NH_3				2.0	0.5		2.5
N_2H_4	x	x		1.5	0.5		4.0
MMH				1.5	0.5		2.0
50-50				2.0	0.5		2.5
RP-1				2.0	0.5		2.5
C_2H_6					0.5		0.5
CH_4				1.0	0.5		1.5
B_2H_6				1.0	0.5		1.5
B_5H_9	x	x		1.5	0.5		4.0
A-5				0.5	0.5		1.0
H_2				0.5	0.5		1.0
HDZP					0.5		0.5
NF_3/NH_3				2.0	0.5		2.5
N_2H_4	x			1.5	0.5		3.0
MMH				1.5	0.5		2.0
RP-1				2.0	0.5		2.5
CH_4				1.0	0.5		1.5
B_2H_6				1.0	0.5		1.5
B_5H_9	x			1.5	0.5		3.0
A-5				0.5	0.5		1.0
HDZP					0.5		0.5

TABLE 17

(Continued)

Propellants		Ignition Tests	Research Thrust Chamber	Thrust Chamber Development	System Development Fuel	System Development Oxidizer	Engine System Development	Experience Rating
OF_2/	NH_3				2.0	1.0		3.0
	N_2H_4				1.5	2.0		3.5
	MMH	x			1.5	2.0		3.5
	50-50	x			2.0	1.0		4.0
	RP-1	x	x		2.0	1.0		5.0
	C_2H_6					1.0		1.0
	CH_4	x	x	x	1.5	1.0		6.5
	B_2H_6	x	x	x	1.0	1.0		6.0
	B_5H_9	x			1.5	1.0		3.5
	A-5				0.5	1.0		1.5
	H_2	x	x		2.0	1.0		5.0
	HDZP					1.0		1.0
	C_3H_8					1.0		1.0
	UDMH					1.0		2.0
	$C_2H_5B_{10}H_{13}$					1.0		1.0
O_2/	NH_3	x	x	x	2.0	2.0	x	10.0
	N_2H_4	x	x	x	2.0	2.0		8.0
	MMH	x	x		1.5	2.0		5.5
	50-50	x	x	x	2.0	2.0		8.0
	RP-1	x	x	x	2.0	2.0	x	10.0
	C_2H_6					2.0		2.0
	CH_4	x	x		1.0	2.0		5.0
	B_2H_6				1.0	2.0		3.0

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TABLE 17

(Continued)

Propellants	Ignition Tests	Research Thrust Chamber	Thrust Chamber Development	System Development Fuel	System Development Oxidizer	Engine System Development	Experience Rating
O_2/ B_5H_9				1.5	2.0		3.5
A-5	x	x		0.5	2.0		4.5
H_2	x	x	x	2.0	2.0	x	10.0
HDZP					2.0		2.0
FLOX/NH_3				2.0	1.0		3.0
N_2H_4				1.5	1.0		2.5
MMH				1.5	1.0		2.5
50-50				2.0	1.0		3.0
RP-1				2.0	1.0		7.0
C_2H_6					1.0		1.0
CH_4				1.5	1.0		4.5
B_2H_6				1.0	1.0		2.0
B_5H_9				1.5	1.0		2.5
A-5					1.0		1.0
H_2				1.0	1.0		2.0
HDZP					1.0		1.0
CLF_3/NH_3				2.0	1.0		3.0
N_2H_4	x	x	x	1.5	1.0		6.5
MMH				1.5	1.0		2.5
50-50	x	x	x	2.0	1.0		7.0
RP-1				2.0	1.0		3.0
C_2H_6					1.0		1.0
CH_4				1.0	1.0		2.0

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TABLE 17

(Continued)

Propellants	Ignition Tests	Research Thrust Chamber	Thrust Chamber Development	System Development Fuel	System Development Oxidizer	Engine System Development	Experience Rating
CLF_3/B_2H_6				1.0	1.0		2.0
B_5H_9	x	x		1.5	1.0		4.5
A-5				0.5	1.0		1.5
H_2				2.0			2.0
HDZP					1.0		1.0
NH_3				2.0			2.0
N_2H_4	x	x		1.5			3.5
MMH				1.5			1.5
50-50				2.0			2.0
RP-1				2.0			2.0
C_2H_8							0
CH_4				1.0			1.0
B_2H_6				1.0			1.0
B_5H_9				1.5			1.5
A-5				0.5			0.5
H_2				2.0			2.0
HDZP							0
N_2O_4/NH_3	x	x		2.0	2.0		6.0
N_2H_4	x	x	x	1.5	2.0		7.5
MMH	x	x	x	1.5	2.0		7.5
50-50	x	x	x	2.0	2.0	x	10.0

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TABLE 17

(Continued)

Propellants	Ignition Tests	Research Thrust Chamber	Thrust Chamber Development	System Development Fuel	System Development Oxidizer	Engine System Development	Experience Rating
N_2O_4/RP-1				2.0	2.0		4.0
C_2H_6					2.0		2.0
CH_4				1.0	2.0		3.0
B_2H_6				1.0	2.0		3.0
B_5H_9				1.5	2.0		3.5
A-5	x	x		0.5	2.0		4.5
H_2				1.5	2.0		3.5
C_3H_8					2.0		2.0

To account for the state-of-the-art variations between propellant classes, a propellant combination was rated by its physical state (i.e., solid, liquid, gel slurry, or powder). The rating method is indicated in Table 18. This rating is arbitrary; however, it does indicate a relative degree of difficulty associated with the transfer method.

No transfer is involved with the use of solid propellants; therefore, the system was not degraded because of their use. Liquid propellants must be transferred; however, these transfer methods are well developed. Use of a liquid propellant degrades the system by one. Slurrys and gels represent a lower degree of transfer method development and were degraded by three. Propellants transferred as powders were the lowest and were degraded by five.

PROPULSION-SYSTEM SIMPLICITY

Items which affect the operational aspects of a space propulsion system were considered under the heading of a system simplicity factor. Considered in this factor were (1) propellant combination hypergolicity, (2) purge requirement, (3) hardware childown requirements, and (4) dual pressurization system requirement. Propellant selection affects these items and contributes to the overall complexity of the propulsion system.

A propellant combination that is hypergolic needs no ignition system since the propellants ignite upon contact with one another. This results in considerable propulsion system simplification which is particularly important where a large number of starts are required. Propellant combinations are designated as hypergolic when ignition occurs within 5-10 milliseconds following contact. Where ignition takes longer, there is danger of building up large amounts of propellant in the combustion chamber resulting in an excessive pressure "spike". These propellant combinations would be classified as non-hypergolic, and some form of ignition device is required.

Most propulsion systems that are designed for restart require that portions of the engine which are downstream of valves be purged of propellant at cutoff. This prevents combustible mixtures of propellants from occuring upstream of the injector which might ignite either spontaneously or at restart. An additional requirement for purging results because propellants that remain in the lines and manifolds are subject to the temperature variations of the engine. Freezing of the propellants could occur for the noncryogencis resulting in possible engine failure. A third requirement for purging occurs if the engine must provide a highly accurate or consistant cutoff impulse. In this case, one of the propellants (usually the oxidizer) would be

TABLE 18

PROPELLANT PHYSICAL STATE FACTOR

Indicates Method of Propellant Transfer from Tank to Thrust Chamber

Example

Oxidizer	Fuel	Additive	Additive Compatibility	Rating
Solid	Solid	Solid	Compatible	10
Liquid	Solid (Hybrid)	Solid	Compatible	8
Liquid	Solid (Powder)	—	—	2
Liquid	Liquid	—	—	6
Liquid	Liquid	Liquid	Compatible	6
Liquid	Liquid	Liquid	Incompatible	4
Liquid	Liquid	Solid	Compatible	4
Liquid	Liquid	Solid	Incompatible	0

purged to provide a consistant operating condition immediately after cutoff. Based upon these concepts, a system of determining the requirements for a purge was established and listed below. A purge system using some inert gas obviously adds to the complexity of the overall engine system.

TABLE 19

REQUIREMENT FOR ENGINE SYSTEM PURGE

Propellant Property		Purge Required	
Oxidizer	Fuel	Oxidizer	Fuel
Cryogenic	Noncryogenic		x
Noncryogenic	Cryogenic	x	
Noncryogenic	Noncryogenic	x	x
Cryogenic	Cryogenic	x	
Hypergolic		x	x

In Table 19 , it was assumed that a consistant cutoff impulse was required. The noncryogenic propellants were arbitrarily assumed to be those with freezing points above -60F. Propellants with freezing points above this temperature may freeze if the engine becomes cold.

There is also the possibility that the engine could be oriented during coasting so that the hardware is relatively warm. In this case, a cryogenic propellant might be heated during the start process to the point that a phase change occured. This may result in undesirable start characteristics. To prevent this, a childown procedure precedes opening of the main propellant valves. A small amount of propellant flows through the system until the hardware is cooled to the desired temperature, then the main start sequence is initiated. If either of the propellants in a combination has a normal boiling point below -60F, a childown was assumed to occur prior to start. The use of a childown system adds to the complexity of the engine system.

In most propellant combinations, where an inert gas such as helium is used for pressurization, both propellants of the combination can use the same pressurization system. In the case of hydrogen, however, helium is apparently not desirable because of the solubility of helium in hydrogen. Therefore, propellant combinations using hydrogen will require a dual pressurization system. Generally, gaseous hydrogen will be used to pressurize the hydrogen tank and helium will be used for the oxidizer. This dual system results in additional system complexity.

The system-simplicity factor was established based upon the above items. In the table below, the weighting factor attached to the items in making the rating is indicated.

TABLE 20

SYSTEM SIMPLICITY COMPARISON FACTOR

Item	Rating
1. Hypergolic	
A. Yes	4
B. No	0
2. Purge Required	
A. One	2
B. Two	0
3. Chilldown Required	
A. Yes	0
B. No	2
4. Pressurization	
A. Dual System	0
B. Single System	3

Considering the physical and operational properties of the various propellant combinations, the system simplicity comparison factors were evaluated. The physical property information is contained in Table 6 . The system simplicity factors are listed in Table 21 .

PROPULSION-SYSTEM SENSITIVITY

Variations in the propulsion system steady-state performance (thrust, mixture ratio, etc.) may occur because of variations in the system operating conditions. Propulsion system calibration occurs at given, nominal operating conditions and deviations from this nominal will affect the performance. The operating condition deviations can occur in the thrust chamber and feed system tolerance, pressure regulator tolerance, and propellant density variation. The propellant combination selected affects this sensitivity only through propellant density variation.

A propulsion-system sensitivity factor was used to indicate the relative variations in propulsion system operation caused by the variation of density with temperature. Large variations in density may lead to large ullage requirements or to the need for a mixture ratio control device. The slope of the density-vs-temperature curve was obtained for each propellant at either the normal boiling point for the cryogenic propellants or 70F for the noncryogenic propellants. The larger value of the two "partials" for a propellant combination was used as a basis of rating the propellant combinations on a "one" to "ten" scale. This rating is given in Fig. 7 . The actual density partials and the resulting ratings are given in Table 22 .

PROPELLANT THERMAL STORAGE IN SPACE

The Apollo mission covers an extended period of time in which the vehicle is exposed to the space environment of the earth-moon system. Therefore, thermal storage is one of the criteria affecting selection of propellant combinations for application in an advanced Apollo.

During the several days of the mission propellants for the three propulsion systems must be thermally protected to prevent: (1) an excessive rise in tank pressure, (2) a propellant from freezing, (3) a large loss of propellant from boiloff. Attitude control of the vehicle can provide some protection during the mission. Insulation of propellant tanks provides the additional protection to prevent a propellant from undergoing a bulk temperature change greater than a predetermined allowable range. Protection by attitude control was an invariant between propellant combinations. Therefore, only insulation weight variations between propellant combinations were

TABLE 21

SYSTEM SIMPLICITY AND LAUNCH STORAGE RATING FACTORS

Oxidizer	Fuel	Hypergolic	System Simplicity Rating	Launch Storage Rating	Oxidizer	Fuel	Hypergolic	System Simplicity Rating	Launch Storage Rating
F_2	N_2H_4	Yes	7	6	O_2	NH_3	No	5	6
	50-50	Yes	7	6		CH_4	No	5	6
	MMH	Yes	7	6		C_2H_6	No	5	4
	UDMH	Yes	7	6		B_2H_6	No	5	6
	Hydyne	Yes	7	6		B_5H_9	No	5	8
	NH_3	Yes	7	4		H_2	No	2	6
	CH_4	Yes	7	4		RP-1	No	5	8
	C_2H_6	Yes	7	2		A-5	No	5	8
	B_2H_6	Yes	7	4		HZP	No	5	8
	B_5H_9	Yes	7	6	ClF_3	N_2H_4	Yes	7	6
	H_2	Yes	4	4		50-50	Yes	7	6
	RP-1	Yes	7	6		MMH	Yes	7	6
	A-5	Yes	7	6		UDMH	Yes	7	6
	HZP	Yes	7	6		Hydyne	Yes	7	6
O_2	N_2H_4	No	5	8		NH_3	Yes	7	4
	50-50	No	5	8		CH_4	Yes	7	4
	MMH	No	5	8		C_2H_6	Yes	7	2
	UDMH	No	5	8		B_2H_6	Yes	7	4
	Hydyne	No	5	8		B_5H_9	Yes	7	6

TABLE 21 (CONT.)

Oxidizer	Fuel	Hypergolic	System Simplicity Rating	Launch Storage Rating
ClF_3	H_2	Yes	4	4
	RP-1	Yes	7	6
	A-5	Yes	7	6
	HZP	Yes	7	6
Comp A	N_2H_4	Yes	7	6
	50-50	Yes	7	6
	MMH	Yes	6	6
	UDMH	Tes	7	6
	Hydyne	Yes	7	6
	NH_3	Yes	7	4
	CH_4	Yes	7	4
	C_2H_6	Yes	7	2
	B_2H_6			
	B_5H_9			
	H_2			
	RP-1	Yes	7	6
	A-5	Yes	7	6
	HZP	Yes	7	6
H_2O_2	N_2H_4	No	5	10
	50-50	No	5	10
	MMH	No	5	10
	UDMH	No	5	10
H_2O_2	Hydyne	No	5	10
	NH_3	No	5	8
	CH_4	No	5	8
	C_2H_6	No	5	6
	B_2H_6	No	5	8
	B_5H_9	No	5	10
	H_2	No	5	8
	RP-1	No	5	10
	A-5	No	5	10
	HZP	No	5	10
N_2O_4	N_2H_4	Yes	7	10
	50-50	Yes	9	10
	MMH	Yes	7	10
	UDMH	Yes	7	10
	Hydyne			
	NH_3	Yes	7	8
	CH_4	No	5	8
	C_2H_6	No	5	6
	B_2H_6	No	5	8
	B_5H_9	No	5	10
	H_2			
	RP-1	No	5	10

TABLE 21 (CONT.)

Oxidizer	Fuel	Hypergolic	System Simplicity Rating	Launch Storage Rating
N_2O_4	A-5	Yes	9	10
	HZP			
OF_2	N_2H_4	Yes	7	4
	50-50	Yes	7	6
	MMH	Yes	7	6
	UDMH	Yes	7	6
	Hydyne			
	NH_3	Yes	7	2
	CH_4	No	5	2
	C_2H_6	No	5	4
	B_2H_6	Yes	7	4
	B_5H_9	Yes	7	6
	H_2	No	2	2
	RP-1	No	5	6
	A-5	Yes	7	6
	HZP	Yes	7	6
FLOX	N_2H_4	Yes	7	6
	50-50	Yes	7	6
	MMH	Yes	7	6
	UDMH	Yes	7	6
	Hydyne			
FLOX	NH_3	Yes	7	4
	CH_4	Yes	7	4
	C_2H_6	Yes	7	2
	B_2H_6	Yes	7	4
	B_5H_9	Yes	7	6
	H_2	No	0	4
	RP-1	Yes	7	6
	A-5	Yes	7	6
	HZP	Yes	7	6
N_2H_4	B_5H_9	No	7	10
MOXIE2A	N_2H_4	No	5	6
	50-50	No	5	6
	MMH	No	5	6
	UDMH	No	5	6
	Hydyne			
	NH_3	No	5	4
	CH_4	No	5	4
	C_2H_6	No	5	2
	B_2H_6	No	5	4
	B_5H_6	No	5	6
	H_2			

TABLE 21 (CONT.)

Oxidizer	Fuel	Hypergolic	System Simplicity Rating	Launch Storage Rating	Oxidizer	Fuel	Hypergolic	System Simplicity Rating	Launch Storage Rating
MOXIE 2A	RP-1	No	5	6	N_2F_4	UDMH	No	5	6
	A-5	No	5	6		Hydyne			
	HZP	No	5	6		NH_3	No	5	4
MON	N_2H_4	Yes	7	6		CH_4	No	5	4
	50-50	Yes	7	6		C_2H_6			
	MMH	Yes	7	6		B_2H_6	No	5	4
	UDMH	Yes	7	6		B_5H_9	No	5	6
	Hydyne					H_2	No	2	4
	NH_3	Yes	7	4		RP-1	No	5	6
	CH_4	No	5	4		A-5	No	5	6
	C_2H_6	No	5	2		HZP	No	5	6
	B_2H_6	No	5	4	NF_3	N_2H_4	No	5	4
	B_5H_9	No	5	6		50-50	No	5	6
	H_2	No	2	4		MMH	No	5	6
	RP-1	No	5	6		UDMH	No	5	6
	A-5	Yes	7	6		Hydyne			
	HZP	Yes	7	6		NH_3	No	5	4
N_2F_4	N_2H_4					CH_4	No	5	4
	50-50	No	5	6		C_2H_6	No	5	2
	MMH	No	5	6		B_2H_6	No	5	4

TABLE 21 (CONT.)

Oxidizer	Fuel	Hypergolic	System Simplicity Rating	Launch Storage Rating
NF_3	B_5H_9	No	5	6
	H_2	No	2	4
	RP-1	No	5	6
	A-5	No	5	6
	HZP	No	5	6

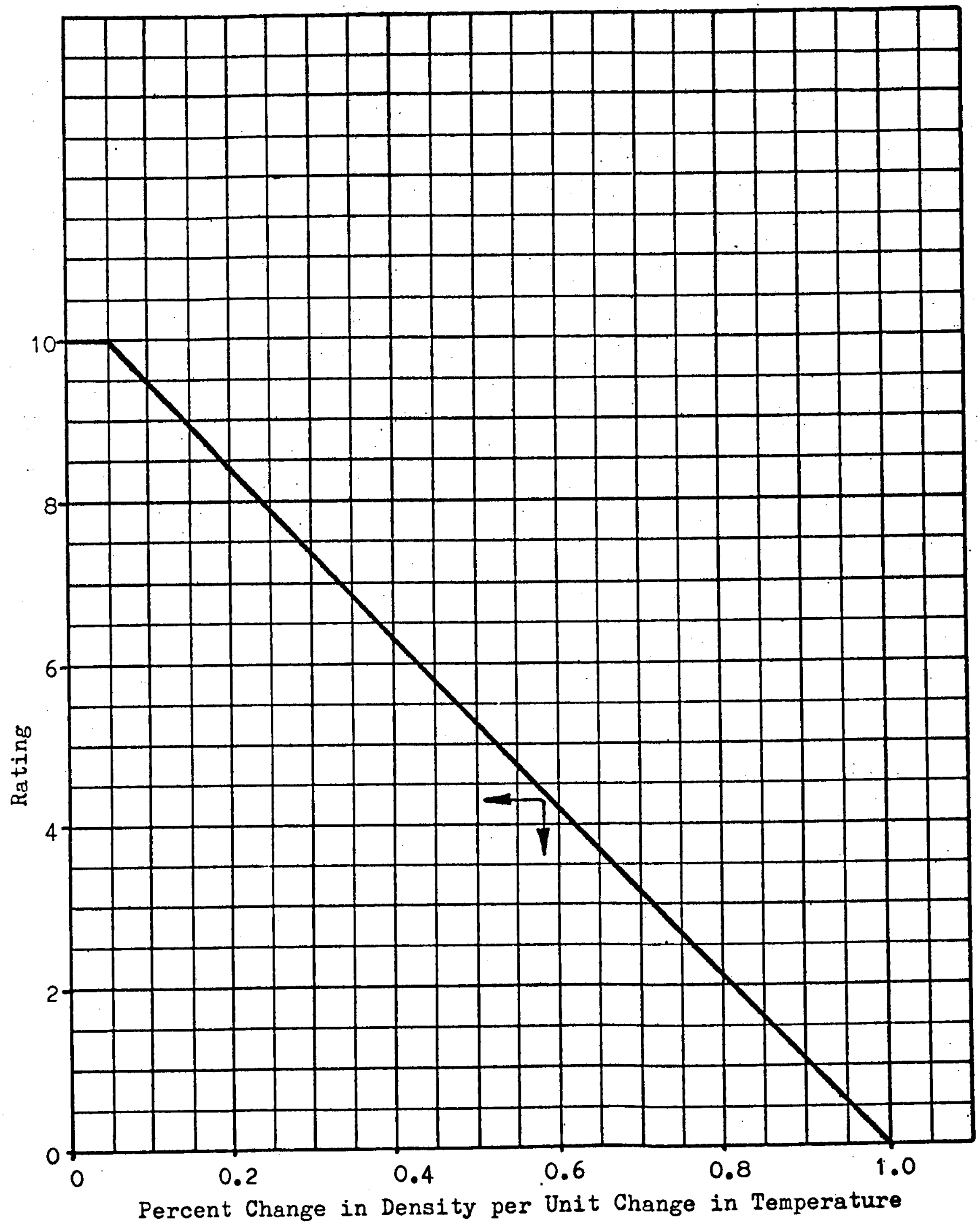

Fig. 7 System Sensitivity Rating Factor

TABLE 22

SYSTEM SENSITIVITY COMPARISON FACTOR

Propellant	Density gm/cc	Temperature, F	Percent $\Delta\rho/\Delta T$	Rating
F_2	1.509	-307.0	-0.262	7.7
OF_2	1.52	-229.0	-0.234	8.1
BrF_5	2.478	70.0	-0.0787	9.7
ClF_3	1.85	53.2	-0.0865	9.6
Comp. A	1.899	6.8	-0.085	9.6
NF_3	1.55	-201.0	-0.201	8.4
N_2F_4	1.66	- 99.4	-0.16	8.9
$FClO_4$	1.695	- 52.3	-0.1091	9.4
MOXIE 2A	1.64	- 60.0	-0.14	9.0
O_2/F_2(10/90)	1.232	-300.0	-0.24	7.9
O_2	1.14	-297.0	-0.2632	7.7
H_2O_2(98)	1.435	70.0	-0.0453	10.0
N_2O_4	1.443	70.0	-0.0832	9.7
HNO_3	1.52	70.0	-0.049	10.0
MON	1.381	68.0	-0.087	9.6
MDFNA	1.63	70.0	-0.072	9.75
RP-1	0.7965	70.0	-0.0502	10.0
CH_4	0.426	-259.0	-0.1883	8.5
C_2H_6	0.547	-127.5	-0.1157	9.4
N_2H_4	1.011	70.0	-0.0460	10.0
MMH	0.875	70.0	-0.0571	10.0
NH_3	0.677	- 20.0	-0.0997	9.5
UDMH	0.79	70.0	-0.0696	9.8

TABLE 22 (CONT.)

Propellant	Density gm/cc	Temperature, F	Percent $\Delta\rho/\Delta T$	Rating
Hydyne	0.08575	70.0	-0.0641	9.9
50-50	0.9	70.0	0.055	9.9
Hydrazoid P	1.095	77.0	-0.05	10.0
B_2H_6	0.438	-135.0	-0.18	8.6
B_5H_9	0.626	70.0	-0.0759	9.7
Hybaline A5	0.736	68.0	-	-
H_2	0.0715	-423.0	-1.0	0

investigated. This weight was used as a rating factor of the propellant combinations. It was merely indicative of the relative degree of difficulty in thermal storage between various propellant combinations.

A factor indicative of the relative amount of insulation was developed based on the following assumptions:

1. Constant reference temperature at outer boundary of insulation
2. Spherical tanks
3. Nonvented storage
4. Maximum propellant bulk temperature corresponding to a vapor pressure of 50 psia
5. Minimum propellant bulk temperature equal to freezing point.

Based on these assumptions, equations were developed for the relative amount of insulation for the fuel and oxidizer tanks (Appendix E).

$$ {}_{o}W_I \propto \left[\frac{1}{\rho_o^{\,2}} \sqrt{\frac{MR(1-e^{-y})}{(1+MR)}} \right]^{2/3} \frac{(T_R - T_o)}{(c_s)_o \, (t_o - T_o)} \qquad (1) $$

$$ {}_{f}W_I \propto \left[\frac{1}{\rho_f^{\,2}} \sqrt{\frac{(1-e^{-y})}{(1+MR)}} \right]^{2/3} \frac{(T_R - T_o)}{(c_s)_f \, (t_o - T_f)} \qquad (2) $$

where

$$ Y = \frac{V_I}{I_s g_o} $$

Equations (1) and (2) are the criteria for the evaluation and comparison of various propellant combinations. The calculated values are proportional to the weight of insulating material that would be required for a mission. A small value indicates an easily-stored propellant while a large value indicates that the propellant is more difficult to store.

An insulation factor was determined by summing the values for the fuel and oxidizer. The factor was based on the physical properties of the propellants, a velocity increment of 7000 fps, and the specific impulses and mixture ratios listed in Table 10 . The reference (or heat source) temperature was assumed to be in the range of -65F to 165F since the propellant tanks of the Apollo are surrounded by auxiliary electronic equipment which is generally maintained at temperatures in this region. In the calculation of the relative insulation weight, the reference temperature was selected to give the largest heat flow estimate.

A rating from one to ten was determined using Fig. 8 . This curve was established from a consideration of the range of values that was obtained from the different propellants. The relative insulation factor for the propellant combinations is presented in Table 23 .

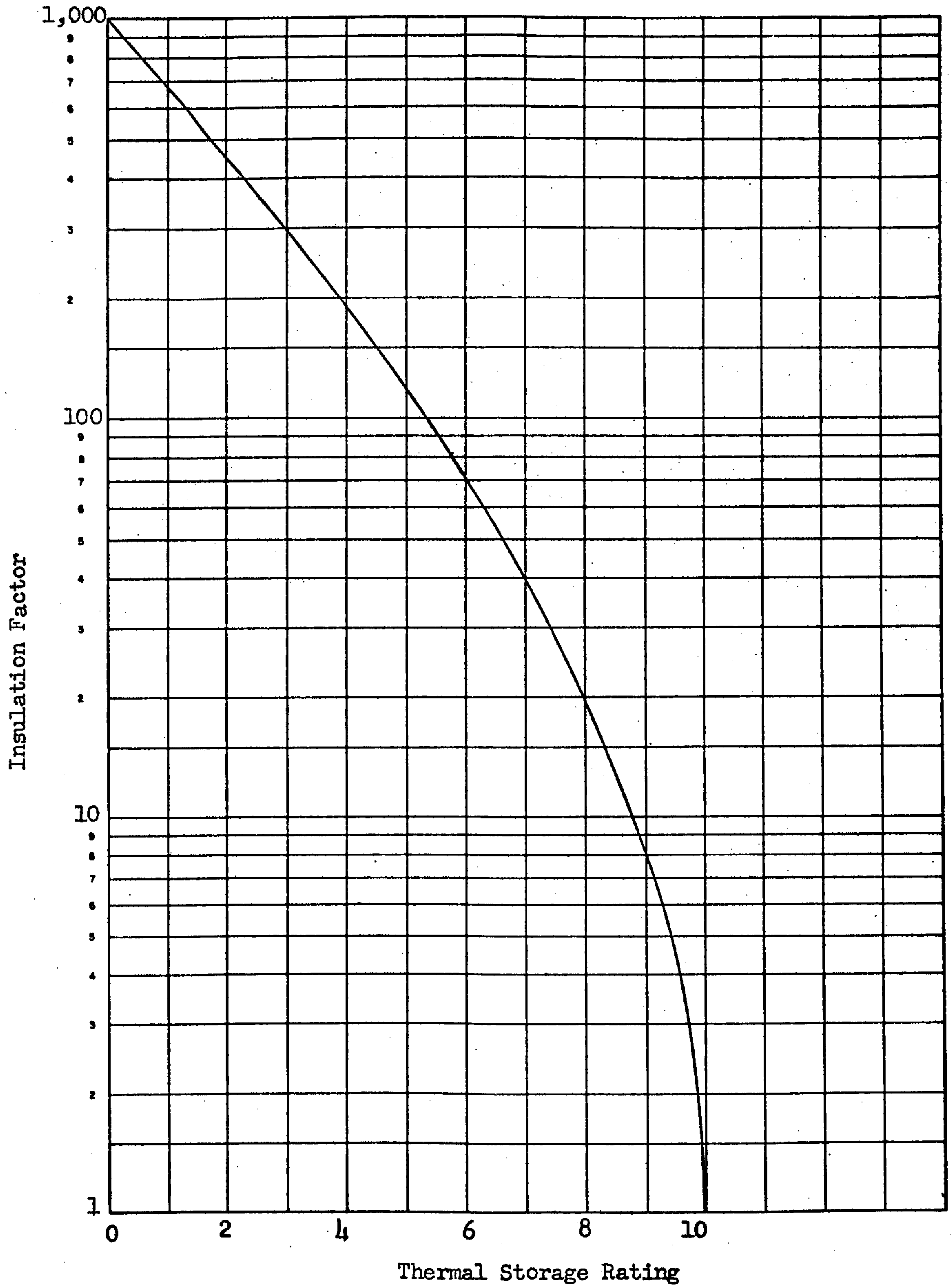

Fig. 8 Insulation Factor vs Thermal Storage Rating for Advanced Apollo Mission.

TABLE 23

PROPELLANT INSULATION FACTOR

Oxidizer	Fuel	Oxidizer Insulation Factor	Fuel Insulation Factor	System Insulation Factor
ClF_3	B_2H_6	2.02	11.6	13.62
	B_5H_9	2.08	1.43	3.51
	Hybaline A-5	2.02	1.14	3.16
	Hydrazoid-P	1.87	0.77	2.64
	H_2	2.02	360.4	362.42
	MMH	1.90	0.92	2.82
	N_2H_4	1.90	3.18	5.08
	N_2H_4 UDMH	2.02	2.43	4.45
	UDMH	1.96	1.31	3.27
ClF_5	B_2H_6	3.19	11.5	14.69
	Hybal A5	3.09	1.11	4.20
	Hydrazoid-P	2.85	0.75	3.60
	MMH	2.95	0.91	3.86
	NH_3	3.09	2.77	5.86
	N_2H_4	2.99	2.85	5.84
	N_2H_4 UDMH	2.95	2.35	5.30

TABLE 23 (CONT.)

Oxidizer	Fuel	Oxidizer Insulation Factor	Fuel Insulation Factor	System Insulation Factor
ClO_3F	B_2H_6	6.06	14.6	20.66
	B_5H_9	6.23	1.72	7.95
	H_2	6.23	434.3	440.53
	MMH	6.14	0.98	7.12
	N_2H_4	5.71	3.18	8.89
FLOX 30-70	B_2H_6	24.27	14.4	38.67
	B_5H_9	24.73	1.76	26.49
	H_2	24.27	455.8	480.07
	NH_3	23.59	3.37	26.96
	N_2H_4	22.44	3.45	25.89
	RP-1	23.36	1.61	24.97
90-10	CH_4	20.68	19.26	39.94
	C_2H_6	20.68	10.25	30.93
	MMH	19.51	0.89	20.40
	N_2H_4 UDMH	19.51	2.35	21.86
	UDMH	19.27	1.40	20.67
F_2	B_2H_6	19.87	11.9	31.77
	B_5H_9	19.39	2.78	22.17

TABLE 23 (CONT.)

Oxidizer	Fuel	Oxidizer Insulation Factor	Fuel Insulation Factor	System Insulation Factor
	CH_4	19.63	19.40	39.03
	C_2H_6	19.63	10.15	29.78
	Hydrazoid-P	17.97	0.77	18.74
	Hydyne	18.92	1.15	20.07
	H_2	19.39	381.9	401.29
	MMH	18.92	0.91	19.83
	NH_3	19.16	2.77	21.93
	N_2H_4	18.45	2.79	21.24
	N_2H_4 UDMH	18.68	2.39	21.07
	RP-1	19.39	1.69	21.08
	UDMH	19.16	1.33	20.49
H_2O_2	B_2H_6	6.43	16.0	22.43
	B_5H_9	6.66	1.93	8.59
	Hybal A5	5.97	1.62	7.59
	H_2	7.19	415.8	422.99
IRFNA	B_5H_9	1.88	1.74	3.62
	Hybal A5	1.56	1.73	3.29
	H_2	1.88	440.4	442.20

TABLE 23 (CONT.)

Oxidizer	Fuel	Oxidizer Insulation Factor	Fuel Insulation Factor	System Insulation Factor
MDFNA	B_5H_9	3.46	1.85	5.31
MON	B_2H_6	3.45	14.6	18.05
	B_5H_9	3.64	1.74	5.38
	Hybal A5	3.53	1.39	4.92
	H_2	2.42	437.4	439.82
	MMH	3.53	0.96	4.49
	N_2H_4	3.33	3.23	6.56
	N_2H_4 UDMH	3.49	2.55	6.04
MOXIE 2A	B_5H_9	4.64	1.33	5.97
	CH_4	4.58	17.49	22.07
	Hydrazoid P	4.23	0.71	4.94
	H_2	4.51	357.3	361.81
	MMH	4.41	0.85	5.26
	NH_3	4.47	2.63	7.10
	N_2H_4	4.35	2.63	6.98
	RP-1	4.52	1.52	6.04
NFO_2	B_5H_9	5.04	1.74	6.78
	N_2H_4	4.79	3.89	8.68

TABLE 23 (CONT.)

Oxidizer	Fuel	Oxidizer Insulation Factor	Fuel Insulation Factor	System Insulation Factor
NF_3	B_5H_9	24.10	1.45	25.55
	H_2	24.04	341.9	365.94
	N_2H_4	22.68	2.79	25.47
	UDMH	23.25	1.27	24.52
N_2F_4	B_2H_6	3.53	11.0	14.53
	B_5H_9	3.62	1.41	5.03
	CH_4	3.57	18.23	21.80
	C_2H_6	3.53	9.37	12.90
	Hydyne	3.43	1.08	4.51
	H_2			
	MMH	3.43	0.86	4.29
	NH_3	3.48	2.70	6.18
	N_2H_4	3.39	2.69	6.08
	N_2H_4 UDMH	3.39	2.23	5.62
	RP-1	3.48	1.57	5.05
	UDMH	3.43	1.26	4.69
N_2H_4	B_5H_9	3.51	2.16	5.67
	Hybal A5	4.06	1.11	5.17

TABLE 23 (CONT.)

Oxidizer	Fuel	Oxidizer Insulation Factor	Fuel Insulation Factor	System Insulation Factor
N_2O_4	B_2H_6	2.80	14.6	17.40
	B_5H_9	2.87	1.76	4.63
	Hybal A5	2.77	1.39	4.16
	Hydrazoid-P	2.58	0.82	3.40
	H_2	2.93	440.4	443.33
	N_2H_4	2.61	3.34	5.95
OF_2	B_2H_6	17.36	13.6	30.96
	B_5H_9	17.58	1.62	19.20
	CH_4	18.23	18.38	36.61
	C_2H_6	17.79	9.27	27.06
	Hydrazoid-P	16.06	0.82	16.88
	Hydyne	17.14	1.10	18.24
	H_2	17.19	415.8	432.99
	MMH	17.36	0.91	18.27
	NH_3	17.14	3.09	20.23
	N_2H_4	16.49	3.07	19.56
	N_2H_4 UDMH	16.93	2.55	19.48
	RP-1	18.01	1.49	19.50
	UDMH	17.14	3.13	20.27

TABLE 23 (CONT.)

Oxidizer	Fuel	Oxidizer Insulation Factor	Fuel Insulation Factor	System Insulation Factor
ONF_3	B_5H_9	4.54	1.47	6.01
	MMH	4.33	0.88	5.21
	NH_3	4.54	3.02	7.61
	N_2H_4	4.26	3.07	7.33
	RP-1	4.62	1.57	6.19
	UDMH	4.54	1.24	5.78
O_2	B_2H_6	26.22	15.5	41.72
	B_5H_9	26.89	1.89	28.78
	CH_4	28.02	21.46	49.48
	Hybal A5	24.86	1.50	26.36
	Hydyne	25.99	1.22	27.21
	H_2	26.58	471.2	497.78
	MMH	25.31	1.04	26.35
	N_2H_4	23.73	3.45	27.18
	N_2H_4 UDMH	25.09	2.78	27.87
	RP-1	27.57	1.71	29.28
	UDMH	25.99	1.49	27.48

TABLE 24

SPACE STORAGE ANALYSIS

NOMENCLATURE

Symbols

Q	Heat Capacity
T	Initial Bulk Temperature
t	Final Bulk Temperature
τ	Time of Propellant Storage
A	Tank Surface Area
A_c	Surface Area of Conductive Path
k	Coefficient of Thermal Conductivity
X	Conductive Path Length
W	Weight
P	Proportion of Propellant that is Boiled Off
C_s	Specific Heat Capacity
ΔH_v	Heat of Vaporization
ρ	Density
ΔV_I	Mission Ideal Velocity Increment
g_o	Gravitational Constant
MR	Weight Mixture Ratio

Subscripts

o	Oxidizer
f	Fuel
R	Reference Environment
I	Insulating Material
S	Support Structure

PROPELLANT TOXICITY

Propellant toxicity can affect many aspects of the development and use of a propulsion system. During development, propellant transport and handling, propulsion system leakage and disposal of exhaust products during testing, all necessitate a different engineering approach for toxic propellants. In the launch operation, the areas of propellant loading, propellant storage, and the possibility of low altitude abort create problems with toxic propellants.

In the rocket industry, there is considerable experience with the testing, development, and launch of toxic propellants. Although the handling and usage of toxic propellants is not a serious problem, the technology is well developed. Propellant toxicity difficulties occur largely in areas where there is little control of propellants, primarily when a large amount of propellant is spilled on the launch pad or there is a low altitude abort. In these cases, propellant vapor could drift downwind for considerable distances, perhaps even reaching densely inhabited areas.

This situation was analyzed using a greatly simplified model described in Reference 1 . An instantaneous mass of propellant vapor was considered and the downwind distance, at which a hazardous concentration of propellant could occur was estimated. Propellant toxicity is usually given in terms of a maximum allowable concentration (MAC) over an extended exposure time (8 hours/day, 40 hours/week). From the MAC, (usually given in parts per million (PPM) on a volume basis) the concentration (gm/m^3) was determined assuming standard temperature and pressure. For short term exposure, an allowable concentration ten times the MAC was assumed as indicated in Reference . A ten-minute exposure time was assumed based upon the ten-minute containment time estimated in the reference. Thus, a maximum allowable dosage was estimated.

The dispersion of the propellants depends upon the climatological conditions. An average wind velocity of 10 mph was assumed as typical of the AMR, and guided by the reference, a lapse temperature gradient of -13 degreesF was assumed. A propellant source strength of 25,000 pounds (approximately 50 percent of the propellant in the Apollo) was assumed.

With the numerous assumptions indicated above, Reference 1 can be used to estimate the downwind hazard distance as a function of the propellant under consideration. Based upon this downwind distance, ratings were assigned. These are indicated in Fig. 9 . In establishing the rating, an AMR location was assumed with the wind blowing from the ocean. It was estimated that there would be no personnel within 0.25 miles. Facilities might be encountered at a distance of 2.0 miles. At a distance of 5-6 miles, "civilian" populations might be exposed; while at a distance greater than 10 miles, densely populated areas might be exposed. With these assumptions in mind, the ratings were assigned.

Using the rating scheme, the toxicity rating factors were assigned to the propellants. For a propellant combination, the lowest rating of the component propellants was used. The MAC values used and the resulting toxicity ratings are given in Table 25 .

PROPELLANT LOGISTICS

The logistics of a propellant provide an indication of the ease in which a propulsion-system development program can be conducted using that propellant. The logistics of a propellant ordinarily consider both the production capacity and the cost. However, studies (Ref. 2) have indicated that propellant cost has little effect upon propellant selection for a space vehicle. Therefore, a rating factor comparing the logistics of the propellants was developed based upon current propellant production rates.

The rating factor was developed for an individual propellant. The rating for a propellant combination (or mixture) was the worst rating of its component propellant ratings. The propellant production rate was divided into various categories which were rated on a 0 to 10 scale. These are described in Table 26 .

From a consideration of the thrust levels of the Apollo propulsion system, a production rate over 100,000 pounds/month was considered not to inhibit a development program. In the case of some propellants, an "unlimited" production capability exists, however, there may be either very large commitments or the production facilities are not currently in use. The case of unrestricted production rates greater than 100,000 pounds/month was given a rating of ten. The restricted cases were given lesser ratings. Lower production capacities were ranked lower in the rating system. Any production capacity less than 10 pounds/month was considered a research quantity. Where the propellant had merely been synthesized, an abrupt decrease in the rating was assigned for a zero rating.

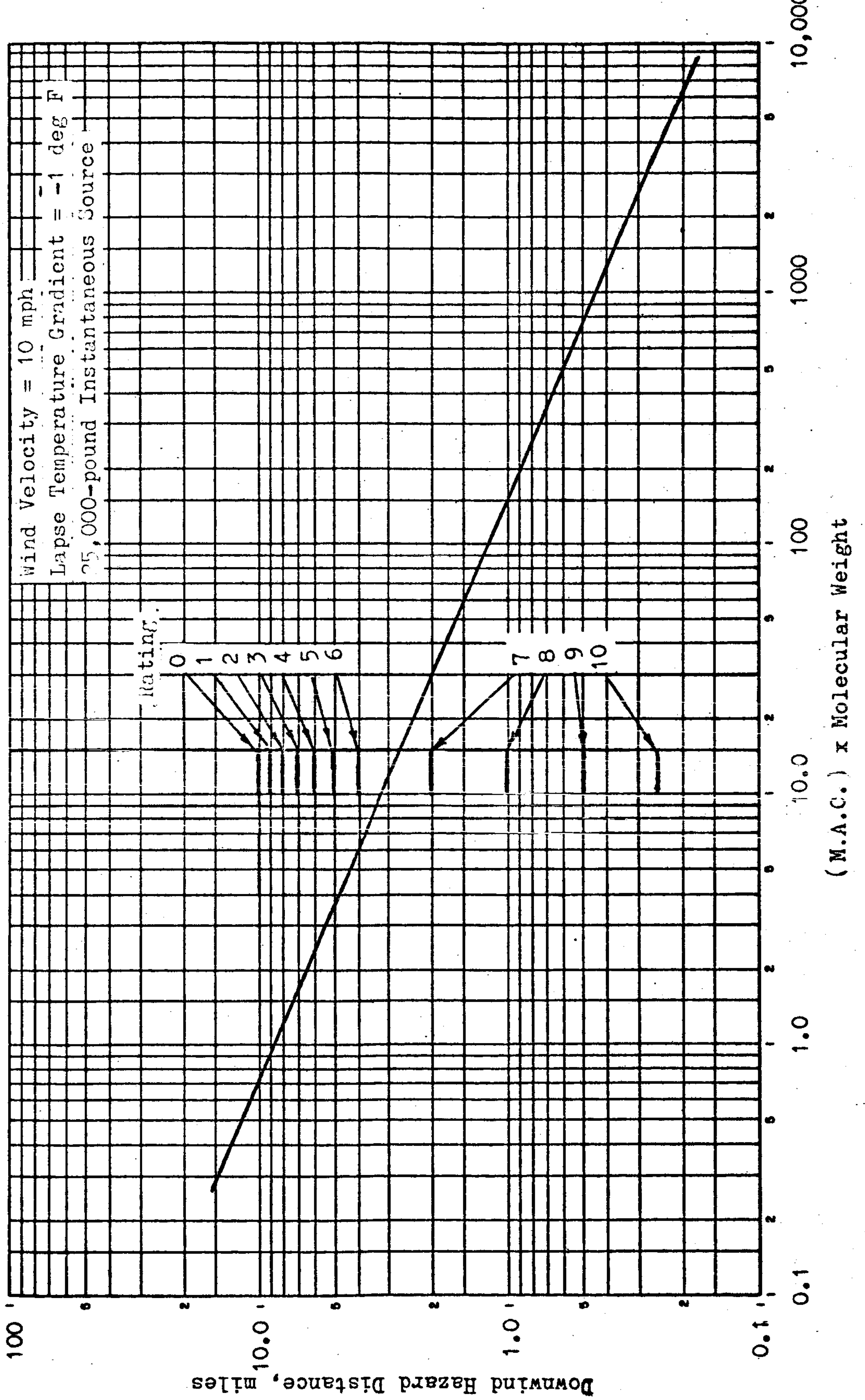

Fig. 9 Toxicity Hazard Rating

TABLE 25

PROPELLANT TOXICITY COMPARISON FACTOR

Propellant	MAC (PPM)	Comparison Factor
1. F_2	0.1	5
2. OF_2	0.005	0
3. NF_2	100	10
4. N_2F_4	~5**	8
5. N_2O_4	5	8
6. MON	5	8
7. Comp A	~0.1**	6
8. BrF_5	3	8
9. ClF_3	0.1	6
10. $FClO_4$	5	8
11. H_2O_2*	1.0	7
12. O_2	---	10
13. MOXIE	~5**	8
14. RP-1	500	10
15. CH_4	90,000	10
16. C_2H_6	50,000	10
17. H_2	---	10
18. B_2H_6	0.1	4
19. B_5H_9	0.005	0
20. Hybaline (A-5)	~500**	10
21. UDMH	0.5	7
22. N_2H_4	1.0	7
23. MMH	1.0	7
24. Hydyne	1.0	7
25. NH_3	100	9
26. Be, BeH_2	2×10^{-6} gms/m^3	0
27. Al	50×10^{6} particles/m^3***	10
28. Li, LiH	25×10^{-6} gm/m^3	0

* Nontoxic, Irritant
** Estimates
*** 50 Micron Particles Assumed

TABLE 26

PROPELLANT LOGISTICS RATING FACTOR

Production Capacity Category	Rating
1. Greater than 100,000 pounds/month; Unrestricted	10
2. Greater than 100,000 pounds/month; Large Commitments	9
3. Greater than 100,000 pounds/month; Upon Demand	8
4. Greater than 10,000 pounds/month	7
5. Greater than 1,000 pounds/month	6
6. Greater than 100 pounds/month	5
7. Greater than 10 pounds/month	4
8. Research Quantity	3
9. Synthesis Only	0

Based upon the firing time required during the three year development of the LEM descent propulsion system, average total propellant requirements were estimated. These are listed in Table 27 .

TABLE 27

ESTIMATED AVERAGE TOTAL PROPELLANT REQUIREMENTS FOR

APOLLO PROPULSION SYSTEM DEVELOPMENT

Total Operation Time, seconds = 82,000
Development Period, years = 3

Module	Thrust, pounds	Total Propellant Requirement pounds/month
Service	22,500	165,000
Descent	10,500	77,000
Ascent	4,000	29,300

The total propellant required includes both propellants: oxidizer and fuel. Individual requirements depend upon mixture ratio.

The current and future propellant availability and the propellant-logistics factor rating are given in Table 28 . The comparison factor was based upon the current availability.

THRUST CHAMBER COOLING

The high temperature associated with the combustion process of the liquid-propellant rocket engine is a major consideration in thrust chamber design. The purpose of this was to evaluate the relative thrust chamber cooling capabilities of the various propellants. There are various methods of protecting the motor walls against the high temperatures. Among the more commonly considered cooling schemes are: ablative liners, regenerative cooling, film or transpiration cooling, and radiation cooling. The applicability of any one of the cooling techniques is strongly dependent on the propellant combination to be used.

TABLE 28

PROPELLANT LOGISTICS

Propellant	Propellant Availability (Approximate) Current	Future	Logistics Rating
1. O_2	unlimited	unlimited	10
2. F_2	100,000 pounds/ month	probably unlimited	10
3. OF_2	1,000 pounds/ month	unlimited*	6
4. NF_3	research quantity	unlimited*	3
5. N_2F_4	research quantity	unlimited*	3
6. N_2O_4	200 tons/day; large commitments	unlimited	9
7. MON	1 ton/day	unlimited	7
8. H_2O_2	unlimited	unlimited	10
9. ClF_3	25 ton/year	unlimited	6
10. BrF_5	limited because of small demand	probably unlimited*	3
11. $FClO_4$	research quantity	unlimited*	3
12. Comp A	30 pounds/ month	unlimited*	4

TABLE 28, CONT.

PROPELLANT LOGISTICS

Propellant	Propellant Availability (Approximate) Current	Future	Logistics Rating
13. MOXIE	50 pounds/ month	unlimited*	4
14. RP-1	unlimited	unlimited	10
15. CH_4	unlimited on demand**	unlimited	8
16. C_2H_6	unlimited on demand**	unlimited	8
17. B_2H_6	25,000 pounds/ month on demand**	unlimited*	7
18. B_5H_9	25,000 pounds/ month on demand**	unlimited*	7
19. UDMH	unlimited; large commitments	unlimited	9
20. N_2H_4	unlimited; large commitments	unlimited	9
21. MMH	300,000 pounds/ month on demand**	unlimited	8
22. NH_3	unlimited	unlimited	10
23. H_2	unlimited	unlimited	10
24. Hybalines	300 pounds/ month	unlimited*	5

TABLE 28 , CONT.

PROPELLANT LOGISTICS

Propellant	Propellant Availability (Approximate) Current	Future	Logistics Rating
25. Al	unlimited	unlimited	10
26. AlH_3	30 pounds/ month	unlimited*	4
27. Be	20,-100,000 pounds/ month	claimed to be unlimited	7
28. BeH_2	research quantity	Based on Be*	3

*2-3 Year Lead Time Required

** Production Very Small; Facilities Exist

Of the mentioned cooling schemes, regenerative and ablative cooling are the most widely used techniques in propulsion systems currently in production or development. Therefore, the relative comparisons in this study will be based upon these two cooling techniques. A simplified analytical factor for the relative comparison of the case of thrust chamber cooling was developed. The operating limitations and properties of the various propellant combinations that are pertinent to the particular cooling technique are used as inputs in the evaluation of this factor.

Regenerative Cooling

Regenerative cooling is accomplished by flowing the coolant through the thrust chamber cooling jacket on its way from the propellant tank to the injector. The performance penalty associated with this cooling system is the additional tank pressure or turbopump weight and power required to overcome the cooling jacket pressure drop.

The criteria for adequate cooling is that the coolant must maintain the thrust-chamber walls at temperatures below that at which failure would occur. Materials commonly used for regenerative thrust chamber walls such as stainless steel, nickel or inconel have temperature limitations in the region of 1500 F to 2000 F. Alternatively, the bulk temperature of a liquid coolant may be limited by the saturation temperature or by the decompostion temperature of the coolant. Many of the propellants that will be considered as the coolant have thermal instability characteristics such that their maximum temperature is considerably below that of the thrust chamber wall material. In such cases, the maximum allowable coolant temperature becomes the system limitation. Experimental data have been used in determining the upper temperature limit for a number of liquid coolants.

In evaluating the relative ease with which a thrust chamber, using certain propellant combinations, can be regeneratively cooled, three basic system characteristics are considered in computing the final regenerative cooling rating factor. These basic characteristics are: the heat flux incident on the thrust chamber wall, the heat capacity of the coolant flow, and the cooling jacket pressure loss encountered in providing adequate cooling capabilities to the thrust chamber design. In view of the extensive list of propellant combinations to be evaluated, a simplified, parameter method was used. The pertinent equations from a detailed analysis were employed and modified by eliminating the parameters that would equal or nearly equal for all systems. The results of this simplified analysis was then used as a relative comparison factor.

The heat transfer rates to a nozzle wall are calculated by the following equation

$$Q/A = h_g \left[T_c - T_{wg}\right]$$

where h_g, the gas-side convection heat-tranfer coefficient for flow through a convergent-divergent rocket nozzle is obtained from the Bartz equation.

$$h_g = \left[\frac{0.026}{D^{*0.2}}\left(\frac{u^{0.2}C_{pc}}{NPr^{0.6}}\right)\left(\frac{P_c g_c}{C^*}\right)^{0.8}\left(\frac{D^*}{r_c}\right)^{0.1}\right]\left(\frac{A^*}{A}\right)^{0.9}\sigma$$

A relative value was obtained for the film coefficient at the throat ($\frac{A^*}{A} = 1$) by assuming chamber pressure and chamber geometry equal for all propellant combinations and eliminating these parameters from the equation. The resulting relationship was:

$$h_g \propto \frac{C_{pc}}{N_{Pr}^{0.6} C^{*0.8}}$$

Assuming that the Prandtl number (N_{Pr}) and thrust coefficient (C_F) do not vary greatly from one propellant combination to another, the expression for the "relative film coefficient" can be further simplified and substituted into the Newton rate equation to yield the relative heat flux values:

$$Q/A \propto \frac{C_{pc}}{I_s^{0.8}}\left[T_c - T_{wg}\right]$$

where T_c is the combustion temperature within the chamber and T_{wg} is the gas side chamber-wall temperature. The gas-side wall temperature was established as the lower of the wall temperature limitation of 1500F (corresponding to the recrystalization limit for stainless steel) or the maximum coolant bulk-temperature limit. The temperature gradient across the tube wall was considered negligible.

Regenerative coolants may also be evaluated on the basis of heat capacity, density and temperature limits. These are the primary factors governing the required coolant velocity and the subsequent pressure drop at a given local heat flux. Either the coolant temperature limitations or the maximum tube wall temperature govern the total amount of heat to be absorbed from inlet to exit of the coolant passage. In most regenerative cooling designs, the fuel is selected as the coolant. The fuels generally have better coolant properties than the oxidizer; another reason is the potential hazard that exists in the event of a tube failure with the oxidizer as the coolant. If the fuel is used as the coolant, and a leak occurs, the fuel will cool the damaged portion of the tube as it flows through the fracture. If the oxidizer were used as the coolant, the leak would result in further damage to the chamber, caused by the heat generated in the reaction of the oxidizer and fuel-rich combustion gases. There may be some propellant combinations where the fuel has poor coolant properties and the oxidizer becomes the better choice. Also, in the case of hybrid systems, the oxidizer offers the only possibility for a regenerative cooling system. For this comparison, however, the fuel will be considered as the coolant and the hybrid systems will be assumed to be ablatively cooled.

The allowable total heat input to the thrust chamber is limited by the capability of the coolant to carry that heat away. The coolant capabilities are defined by the coolant flowrate, heat capacity, and temperature limitations. The coolant heat-absorbtion capabilities can best be represented by the following relationship.

$$\dot{Q}_T = \dot{W}_F C_p [T_F - T_i]$$

$\dot{W}_F$ = fuel flowrate

C_p = Specific heat capacity of the fuel

Substituting for the fuel flowrate results in a more general equation.

$$\dot{Q}_T/F = \frac{1}{(1+MR)I_s} \left[C_p(T_f - T_n)\right]$$

(The specific impulse and mixture ratio values used in the comparison are those corresponding to the maximum vacuum specific impulse at 1000 psi chamber pressure.)

The coolant bulk temperature at any point in the coolant passage is simply the sum of the inlet bulk temperature (T_i) and the temperature rise to that point. For this comparison, the initial temperature (T_i) is assumed as the lower of either the normal boiling point of the coolant or 70F. The final temperature (T_F) was taken as the lower of either the maximum allowable coolant bulk temperature or the material temperature limit (1500F).

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Since the parameter $(\frac{\dot{Q}_T}{F})$ resulting from this equation is representative of the amount of heat that can be absorbed by the coolant as it flows through the jacket, it is desirable to have as high a value as possible. For the previously described factor $(\dot{Q}/A)$, representing the relative heat flux incident on the chamber walls, it is desirable to have as low a value as possible. These two factors are related insofar as the coolant capabilities of the coolant system must be compatible with the incident heat flux resulting from a particular propellant system. This suggests the possibility of combining these two parameters into a single factor that can be used to rate the various propellant combinations.

The quotient of the relative heat flux value and the coolant heat capacity $(\dot{Q}/A/\dot{Q}_T/F)$ offers a single parameter on which various propellant combinations can be rated relatively as to their regenerative-cooling capabilities.

Consideration of cooling-jacket pressure drop will also provide an additional parameter for the comparison. The relative factors discussed so far account for the heat flux produced by the combustion process and the absorbtion capabilities of the coolant. One more factor is required to link these together. The heat absorbtion capabilities defined by the factor $(\dot{Q}_T/F)$ are a measure of the total heat rate that can be absorbed by a coolant but no consideration is given to the heat transfer capabilities between the tube wall and the coolant or to the total expected heat-rejection rate from the combustion gases to the motor wall. A principle parameter in determining the feasibility of regenerative cooling is the cooling-jacket pressure drop. This parameter provides a convenient relationship between the incident heat flux and coolant transport properties.

An extremly simplified analytical technique was developed to provide an additional relative comparison. This analysis is described in Appendix F . The resulting relation for the pressure drop is

$$\Delta P \propto \frac{\left[\frac{\dot{Q}/A}{(T_{wc} - T_B)C_p}\right]^{2.5}}{\rho_c}$$

The $\dot{Q}/A$ value used in this equation for the comparison is the relative value previously described and calculated subject to the various simplifying assumptions. The coolant-side wall temperature (T_{WC}) was estimated based upon experimental data for each individual coolant. The coolant bulk temperature (T_B) was taken as the average between the coolant inlet temperature and its maximum allowable temperature. The

ΔP values obtained are again not intended as the actual expected values but merely as a relative number on which the various coolants can be compared.

A review of previous heat-transfer studies and experimental results have indicated those certain limitations and inherent characteristics which are peculiar to the individual coolants and propellant combinations; and where possible, this information was used in generating these overall comparisons. It has been found for the RP-1 fueled systems that a carbon layer deposits on the combustion chamber wall. This layer increases the thermal resistance to heat transfer to the wall. This feature is beneficial in reducing the coolant flow requirements. However, if the wall temperature exceeds 800F the carbon layer was found not to exist thus, increasing the heat flux to the coolant. In Ref. (3), heat transfer studies are mentioned that have shown B_5H_9 to be a poor regenerative coolant since large solid deposits appear on the coolant side of the tube walls at high heat flux values. The deposits will restrict the heat transfer between the tube wall and the coolant. Experimental data have indicated that hydrazine used as a coolant may decompose exothermally at a bulk temperature of approximately 300F depending on the pressure. These are typical of the experimental data that served as inputs to the comparison. A detailed description of the limiting criteria used for the numerous systems will not be attempted because this would become an extensive list in itself. The limiting values used in the calculations are presented along with the results for each of the propellant calculations.

To facilitate the final regenerative cooling comparisons for the large number of propellant combinations to be compared, a rating scale was established for the $\dot{Q}/A/\dot{Q}_T/F$ and ΔP values obtained. The rating factors for these two parameters are presented in Table 29 .

TABLE 29

PROPELLANT RATING FACTORS FOR REGENERATIVELY

COOLED SYSTEMS

ΔP Value	Rating Factor	$Q/A/Q_T/F$ Value	Rating Factor
1 - 10	0	500 - 1000	0
0.1 - 1	1	100 - 500	1
0.01 - 0.1	2	50 - 100	2
0.001 - 0.01	3	20 - 50	3
0.0001 - 0.001	4	10 - 20	4
0.00001 - 0.0001	5	0 - 10	5

The sum of the rating factors for the two parameters then forms the final overall regenerative cooling, propellant rating. This allows the regenerative cooling rating for a particular propellant combination to vary between zero and 10.

For most of the propellant combinations considered, the combustion temperature at the throat and the specific heat capacity of the products of combustion (based on a frozen performance assumptions) were obtained from results of the propellant performance computer program. These values were obtained for chamber pressures of 1000 psi and at the weight mixture ratio for maximum specific impulse. For certain propellant combinations for which the computer results were not available at the desired conditions, the required values were estimated based on established trends. The estimated values are noted in the tables of results.

Ablative Cooling

Thrust chamber cooling by ablation is accomplished by the heat absorbed through melting, vaporization, and sublimation of an ablative material which lines the chamber walls. The most common ablation materials used in rocket thrust chambers are the reinforced plastics. These materials consist of a resin base reinforced with a fibrous material. Of the possible combinations of resins and fibers, phenolic resin impregnated in nylon or Refrasil cloth exhibit the most desirable characteristics for rocket-engine applications.

One phenomenon occurring with ablation materials is char-layer erosion caused by chemical reaction of the char layer with the combustion gases and also by the shearing forces of these high-velocity exhaust gases and solid particles. The errosion process is primarily a function of the combustion temperature and the constituents of the exhaust products.

The ablation process takes place through a complex mechanism and it is extremely difficult to predict the behavior under varying conditions. Limited test results have provided valuable information leading to generalizations in certain areas. However, to provide detailed ablation material requirements for a particular propellant combination and chamber conditions where data are not already available, a test program would provide the best basis for verification of the design.

No attempt will be made to obtain values of ablative thickness requirements for a particular system. This comparison is intended merely to indicate the relative ease by which a propulsion system utilizing the various propellant combinations can be ablatively cooled. Typical test results used in the generalized comparison are presented in Ref (4). The tests were conducted for a Refrasil-phenolic ablator in the O_2/RP-1 and F_2/H_2 experimental rocket engines. Completely different behavior occurred in the two tests. In the former test, the Refrasil melted and a steady-state char layer developed after a few seconds. In the latter test, the silica apparently vaporized because no melt could be detected. The reason for the difference can be briefly explained as follows: In the O_2/RP-1 test, the carbonaceous char layer was chemically attacked by the H_2O vapor in the exhaust gases, leaving the Refrasil exposed which subsequently melts and runs off exposing more carbon. In the F_2/H_2 test, the carbon was inert to the propellant products of combustion. The silica however, melted and vaporized but no char erosion was experienced.

From these and similar results, the theory was held that any propellant combination that produced water vapor in the exhaust gases will have detrimental effects on the ablation process. This theory was used as one of the criteria in the ablative cooling relative ratings. Solid particles in the exhaust products were also found to have damaging effects, because they impinged on the ablative liners.

The rating scale established for this cooling method, which reflects combustion temperature and exhaust-gas constituent effects, is presented in Table 30 , and the following discussion.

TABLE 30

ABLATIVE COOLING COMPARATIVE PROPELLANT

RATING FACTORS

Combustion Temperature Range Deg F	Combustion Temperature Relative Rating Factor
3500-4500	5
4500-5500	4
5500-6500	3
6500-7500	2
7500-8500	1

The corresponding rating factor is assigned according to the combustion temperature of the individual propellant combination. The same combustion temperatures were used as those in the regenerative cooling ratings. If the exhaust gases do not contain water vapor, a value of 3 is added to the temperature rating factor. If the exhaust products do not include solid particals, an additional value of 2 is added to the overall rating factor. This results in possible ablative-cooling comparative propellant-rating factors from 1 to 10, where a rating of 10 reflects the best possible conditions for the ablative cooling method.

The thrust chamber cooling analysis is summarized in Table 31 . For the final selection of a propellant rating factor for the thrust chamber cooling consideration, the rating value for each propellant combination was selected on the basis of the best cooling method (the highest rating factor value between the regenerative and ablative cooling systems).

Detailed heat transfer studies have been conducted previous to this effort for some of the more commonly considered propellants. The results of these studies have been compared with this generalized comparison and it was found that there is close agreement in the results on a relative basis. However, there are many system variations possible that could improve the cooling capabilities of a particular propellant combination. Although the final evaluation of the merits of a particular coolant must await a detailed design and heat transfer analysis, some measure of comparison between propellants can be achieved from the results obtained in this effort.

PROPELLANT STORAGE AT LAUNCH

In launching a space vehicle for missions like that of the Apollo, it is possible that long times may be spent on the launch pad during the countdown. During this countdown and any associated "hold times", the propellant must be maintained in a useable condition. If the propellants are noncryogenic "earth storables", this is no problem. However, use of cryogenics may necessitate insulation schemes or the use of venting, refrigeration and replenishing systems. These add complexity to the launch operation.

A nominal launch ambient environment temperature of 70F is assumed. Propellants which have a usable temperature range including the nominal point will be considered to be inherently "storable on the launch pad". Nontoxic propellants which have a usable range lower than the

TABLE 31

THRUST CHAMBER COOLING

Oxidizer	Fuel	MR	T_c Comb. Temp. F	Max. Coolant Temp. Tm F	Wall Temp. T_{wg} F	Q/A	Q_T/F	Q/Q
ClF_5	A_5	5.0	7480	550	650	31.0	0.140	221
	B_2H_6	7.0	7640	100	200	23.0	0.0505	455
	CH_4	3.0	6240	1500	1500	18.5	0.109	169
	Hydrazoid-P	2.0	6890	300	400	25.0	0.138	181.
	MMH	2.7	6360	300	400	21.9	0.126	174.
	NH_3	3.8	5640	1500	1500	15.5	0.986	15.
	N_2H_4	2.6	6620	300	400	21.4	0.129	165
	50-50	2.72	6502	500	600	21.2	0.225	94.
B_5H_9	N_2H_4	1.3	4552	300	400	35.2	0.188	187.
ClF_3	A5	5.0	6952	550	650	30.0	0.148	203.
	B_2H_6	7.0	7090	100	200	21.6	0.0541	400.
	B_5H_9	6.7	8182	200	300	27.0	0.0276	980.
	CH_4	7.95	4802	1500	1500	20.2	0.50	40.
	MMH	2.7	5982	300	400	19.9	0.131	152.
	N_2H_4	2.9	5652	300	400	18.5	0.129	143.
	50-50	2.9	6012	500	600	19.2	0.229	83.
ClO_3F	B_2H_6	3.0	6260	100	200	25.4	0.105	242.
	B_5H_9	4.0	7582	200	300	28.0	0.0407	688.
	MMH	2.24	6040	300	400	24.1	0.148	163.
	N_2H_2	1.4	5782	300	400	23.9	0.208	115.
	UDMH	2.7	5202	500	600	17.4	0.225	77.

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ANALYSIS

/A /F	ΔP	ΔP Rating	Q/A / Q/F Rating	Sum	Ablative Rating	Final Rating
.0	0.0113	2	1	3	5	5
.0	0.0227	2	1	3	4	4
.0	0.000228	4	1	5	8	8
.0	0.0131	2	1	3	2	3
.0	0.00933	3	1	4	8	8
.7	0.000072	5	4	9	8	9
.0	0.0071	3	1	4	7	7
.2	0.00294	3	2	5	8	8
.0	0.0242	2	1	3	7	7
.0	0.0104	2	1	3	5	5
.0	0.00910	2	1	3	5	5
.0	0.0710	2	0	2	4	4
.4	0.000284	4	3	7	6	6
.0	0.00721	3	1	4	8	8
.0	0.00492	3	1	4	8	8
.8	0.00259	3	2	5	8	8
.0	0.0297	2	1	3	3	3
.0	0.0772	2	0	2	1	1
.0	0.0116	2	1	3	5	5
0	0.00876	3	1	4	5	5
4	0.00273	3	2	5	6	6

TA

THRUST CHAMBER

Oxidizer	Fuel	rm	T_c Comb. Temp. F	Max. Coolant Temp. Tm F	Wall Temp. T_{wg} F	Q/A	Q_T
FLOX (30-70)	B_2H_6	2.9	6640	100	200	23.9	0.09
	B_5H_9	3.2	7540	200	300	25.0	0.45
	H_2	4.4	5030	1500	1500	19.3	1.77
	NH_3	1.8	4220	1500	1500	11.1	1.62
	N_2H_4	1.2	4760	300	400	16.5	0.2
FLOX (90-10)	CH_4	4.7	7190	1500	1500	21.6	0.63
	C_2H_6	3.8	7280	1500	1500	21.6	0.51
	MMH	2.65	6640	300	400	22.0	0.16
	UDMH	1.19	6380	500	600	24.8	0.33
	50-50	2.59	7972	500	600	24.5	0.19
F_2	B_2H_6	5.6	7902	100	200	25.0	0.05
	B_5H_9	4.6	8032	200	300	23.4	0.05
	CH_4	4.5	6952	1500	1500	20.6	0.78
	C_2H_6	3.7	6812	1500	1500	27.2	0.53
	Hydrazoid-P	1.85	7940	300	400	25.6	0.12
	H_2	8.0	6312	1500	1500	24.5	1.01
	MMH	2.48	6500	300	400	21.8	0.11
	NH_3	3.3	7372	1500	1500	20.2	0.90

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BLE 31

COOLING ANALYSIS

/F	$\frac{Q/A}{Q/F}$	ΔP	ΔP Rating	$\frac{Q/A}{Q/F}$ Rating	Sum	Ablative Rating	Final Rating
48	252.0	0.0254	2	1	3	2	3
9	54.5	0.0580	2	2	4	2	4
	10.9	0.0000992	5	4	9	6	9
	6.85	0.0000314	5	5	10	5	10
8	79.2	0.00355	3	2	5	7	7
6	34.0	0.000188	4	3	7	4	7
	42.2	0.000927	4	3	7	4	7
1	13.7	0.00922	3	4	7	4	7
5	74.0	0.00326	3	2	5	5	5
6	125.0	0.00478	3	1	4	3	4
30	470.0	0.0285	2	1	3	4	4
52	423.0	0.106	1	1	2	4	4
3	26.3	0.000875	4	3	7	7	7
7	50.7	0.00136	3	3	6	7	7
5	205.0	0.0139	2	1	3	3	3
	24.2	0.000203	4	3	7	8	8
6	188.0	0.00897	3	1	4	7	7
7	22.3	0.00014	4	3	7	7	7

TABLE 31

THRUST CHAMBER COOLING

Oxidizer	Fuel	MR	T_c Comb. Temp. F	Max. Coolant Temp. Tm F	Wall Temp. T_{wg} F	Q/A	Q_T/F	Q/ Q/
F_2(Con't)	N_2H_4	2.3	7552	300	400	24.6	0.122	202.
	RP-1	2.6	6832	800	800	26.4	0.268	98.
	UDMH	2.5	6342	500	600	21.2	0.202	105.
	50-50	2.4	7652	500	600	24.8	0.209	119.
F_2/BeH_2	MMH	3.35	8290	300	400	27.9	0.0910	306.
H_2O_2	B_2H_6	1.9	4072	100	200	26.0	0.134	194.
	B_5H_9	2.4	4862	200	300	28.0	0.104	269.
	CH_4	7.95	4802	1500	1500	20.2	0.50	40.
	NH_3	2.89	4272	1500	1500	17.7	1.34	13.
	N_2H_4	2.17	4802	300	400	26.4	0.164	161.
	UDMH	3.5	4892	500	600	25.0	0.191	132.
MON	B_5H_9	3.45	6570	200	300	27.6	0.0477	578.
	MMH	2.22	5462	300	400	22.8	0.147	155.
	50-50	2.1	5660	500	600	21.6	0.284	76
MOXIE	B_5H_9	8.2	7820	200	300	27.4	0.0255	107
	CH_4	7.12	6610	1500	1500	19.1	0.516	3
	Hydrazoid-P	2.71	7240	300	400	23.5	0.109	215
	MMH	3.7	7140	300	400	24.1	0.0954	253
	NH_3	4.82	6850	1500	1500	19.3	0.785	24
	N_2H_4	3.3	6920	300	400	24.2	0.112	216

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ANALYSIS

ΔP	ΔP Rating	Q/A / Q/F Rating	Sum	Ablative Rating	Final Rating
0.00954	3	1	4	6	6
0.0129	2	2	4	7	7
0.00442	3	1	4	8	8
0.00487	3	1	4	6	6
0.0168	2	1	3	4	4
0.0314	2	1	3	5	5
0.19	1	1	2	4	4
0.000284	4	3	7	6	7
0.0000980	5	4	9	7	9
0.0110	2	1	3	6	6
0.0205	2	1	3	6	6
0.0740	2	0	2	2	2
0.0102	2	1	3	6	6
0.00173	3	2	5	8	8
0.0735	2	0	2	4	4
0.000243	4	3	7	4	7
0.0128	2	1	3	3	3
0.0116	2	1	3	4	4
0.000121	4	3	7	4	7
0.00910	3	1	4	7	7

TABL

THRUST CHAMBER C

Oxidizer	Fuel	MR	T_c Comb. Temp. F	Max. Coolant Temp. Tm F	Wall Temp. T_{wg} F	Q/A	Q_T/
NFO_2	B_5H_9	3.5	7000	200	300	25.0	0.047
	N_2H_4	1.4	5640	300	400	15.4	0.208
NF_3	B_5H_9	6.7	8182	200	300	27.0	0.027
	H_2	13.3	6332	1500	1500	20.1	0.750
	UDMH N_2H_4	3.16	6672	500	600	24.8	0.189
	N_2H_4	2.7	7182	300	400	24.0	0.123
NTO	A-5	2.2	5980	550	650	27.3	0.271
	UDMH	2.75	5240	500	600	20.0	0.184
N_2F_4	B_2H_6	6.5	7782	100	200	27.6	0.056
	B_5H_9	7.3	8000	200	300	28.3	0.023
	CH_4	6.18	6350	1500	1500	18.8	0.571
	C_2H_6	5.02	5932	1500	1500	15.4	0.454
	H_2	12.0	5530	1500	1500	19.5	0.800
	MMH	3.25	5850	300	400	21.9	0.103
	NH3	4.0	5842	1500	1500	17.4	0.885
	N_2H_4	3.06	7475	300	400	25.0	0.059
	RP-1	3.5	5710	800	800	19.3	0.213
	UDMH	3.1	5780	500	600	28.0	0.187
	50-50	3.3	7192	500	600	25.6	0.182
N_2O_4	B_2H_6	2.85	6030	100	200	22.0	0.109
	B_5H_9	3.35	6532	200	300	27.7	0.087

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OOLING ANALYSIS

	$\frac{Q/A}{Q/F}$	ΔP	ΔP Rating	$\frac{Q/A}{Q/F}$ Rating	Sum	Ablative Rating	Final Rating
4	527.0	0.059	2	0	2	2	2
	74.1	0.003	3	2	5	5	5
6	980.0	0.0710	2	0	2	4	4
	26.8	0.000126	4	3	7	8	8
	131.0	0.00672	3	1	4	7	7
	195.0	0.00897	3	1	4	7	7
	100.0	0.00815	3	1	4	3	4
	109.0	0.00340	3	1	4	6	6
2	490.0	0.0358	2	1	3	4	4
4	1210.0	0.079	2	0	2	4	4
	33.0	0.000232	4	3	7	8	8
	34.0	0.000328	4	3	7	8	8
	24.4	0.000106	4	3	7	8	8
	212.0	0.00912	3	1	4	8	8
	19.7	0.000098	5	4	9	8	9
	424.0	0.0480	2	1	3	7	7
	90.7	0.00584	3	2	5	8	8
	150.0	0.0088	3	1	4	8	8
	141.0	0.00531	3	1	4	7	7
	202.0	0.0206	2	1	3	3	3
	410.0	0.185	1	1	2	2	2

TABLE 31

THRUST CHAMBER COOLING

Oxidizer	Fuel	MR	T_c Comb. Temp. F	Max. Coolant Temp. Tm F	Wall Temp. T_{wg} F	Q/A	Q_T/F	Q/Q
N_2O_4	MMH	2.19	5422	300	400	22.6	0.148	154
	NH_3	2.0	-	1500	1500	3.77	1.76	2
	N_2H_4	1.4	5074	300	400	22.0	0.206	107
	50-50	2.1	5382	500	600	21.2	0.280	75
N_2O_4/BeH_2	MMH	1.03	5430	300	400	19.2	0.209	91
OF_2	B_2H_6	3.6	7442	100	200	28.5	0.0765	373
	B_5H_9	4.0	7852	200	300	25.8	0.0625	413
	CH_4	5.6	6962	1500	1500	19.4	0.653	29
	$C_2H_5B_{10}H_{13}$	3.8	8540	400	500	22.7	0.083	274
	C_2H_6	4.9	7132	1500	1500	19.1	0.407	47
	Hydrazoid-P	1.36	6790	300	400	25.2	0.162	155
	H_2	6.0	5632	1500	1500	25.8	1.33	19
	MMH	2.5	6852	300	400	23.2	0.114	203
	NH_3	2.3	5610	1500	1500	16.3	1.24	13
	N_2H_4	1.6	6412	300	400	23.8	0.163	146
	RP-1	3.8	7352	800	800	16.2	0.18[illegible]	85
	UDMH	2.7	7532	500	600	25.2	0.183	134
	50-50	2.14	7532	500	600	26.0	0.235	111
OF_2/Be	N_2H_4	1.24	6740	300	400	26.8	0.191	140

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ANALYSIS

/A /F	ΔP	ΔP Rating	Q/A / Q/F Rating	Sum	Ablative Rating	Final Rating
.0	0.0100	2	1	3	6	6
.14	0.00000212	5	5	10	7	10
.0	0.00730	3	1	4	6	6
.7	0.00294	3	2	5	6	6
.8	0.00656	3	2	5	4	5
.0	0.176	1	1	2	2	2
.0	0.156	1	1	2	1	2
.7	0.000757	4	3	7	4	7
0	0.2	1	1	2	1	2
0	0.000557	4	3	7	4	7
0	0.0133	2	1	3	4	4
4	0.000108	4	4	8	5	8
0	0.0105	2	1	3	4	4
1	0.0000822	5	4	9	5	9
0	0.0087	3	1	4	5	5
6	0.00472	3	2	5	3	5
0	0.00672	3	1	4	3	4
0	0.00545	3	1	4	3	4
0	0.0116	2	1	3	2	3

TABLE

THRUST CHAMBER CO

Oxidizer	Fuel	MR	T_c Comb. Temp. F	Max. Coolant Temp. Tm F	Wall Temp. T_{wg} F	Q/A	Q_T/F
ONF_3	B_5H_9	6.0	8200	200	300	25.8	0.276
	MMH	3.0	6970	300	400	24.2	0.109
	NH_3	3.0	5280	1500	1500	15.8	1.19
	N_2H_4	2.0	5800	300	400	20.4	0.155
	RP-1	4.0	5560	800	800	21.4	0.212
	UDMH	3.8	5840	500	600	25.7	0.180
O_2	B_2H_6	2.15	6372	100	200	35.0	0.126
	B_5H_9	2.4	7192	200	300	32.0	0.102
	CH_4	3.35	5882	1500	1500	21.4	0.940
	C_2H_6	3.0	6052	1500	1500	21.2	0.668
	H_2	4.5	5612	1500	1500	26.8	1.72
	N_2H_4	0.9	4912	300	400	22.0	0.521
	RP-1	2.6	5842	800	800	22.0	0.257
	UDMH	1.67	5132	500	600	23.9	0.293
	50-50	1.29	5882	500	600	25.4	0.350

31

OOLING ANALYSIS

$\frac{Q/A}{Q/F}$	ΔP	ΔP Rating	$\frac{Q/A}{Q/F}$ Rating	Sum	Ablative Rating	Final Rating
0 93.5	0.089	2	2	4	2	4
222.0	0.0116	2	1	3	4	4
13.2	0.000072	5	4	9	6	9
131.0	0.006	3	1	4	5	5
101.0	0.0075	3	1	4	5	5
143.0	0.00716	3	1	41	5	5
278.0	0.0653	2	1	3	3	3
314.0	0.276	1	1	2	2	2
22.8	0.000325	4	3	7	5	7
31.7	0.000732	4	3	7	5	7
15.5	0.0000608	5	4	9	5	9
42.2	0.00712	3	3	6	6	6
85.5	0.0815	2	2	4	5	5
81.6	0.00630	3	2	5	6	6
72.6	0.00519	3	2	5	5	5

TABLE 32

THRUST CHAMBER COOLING

NOMENCLATURE

Q/A - Heat flux
h_g - Gas side heat transfer coefficient
T_c - Combustion temperature
m - Dynamic viscosity
C_{P_c} - Specific heat capability of products of combustion
P_c - Chamber Pressure
r_c - Radius of curvature of nozzle throat
A - Area
D - Diameter
C^* - Characteristic velocity
T_{wg} - Gas side wall temperature
I_s - Specific impulse
M.R. - Propellant mixture ratio
F - Thrust
T_B - Coolant bulk temperature
N_u - Nusselt number
N_{pr} - Prandtl number
N_{Re} - Reynolds number
C_p - Specific heat capacity of the fuel
T_{wc} - Coolant side wall temperature
K - Thermal Conductivity
h_c - Coolant side heat transfer coefficient
ρ - Coolant density
V - Coolant velocity

nominal temperature and toxic propellants having a molecular weight less than that of air can be vented and the propellant replenished. Toxic propellants, with high molecular weights having a usable range lower than nominal, will need a closed refrigeration system. Using this approach, the ratings below were assigned on the basis of a maximum possible of ten. Solid propellant components were all assumed compatible with the launch environment and to be "storable".

TABLE 33

LAUNCH STORAGE METHOD RATING

Propellant Characteristic	Toxic	Molecular Weight Less Than 29 or Nontoxic
Nominal Temperature Included in Usable Range	5	5
Nominal Temperature not Included in Usable Range	1	3

Propellant physical property information was obtained from Table 6 and toxicity information from Table 25. Using this data, the launch storage-method comparison factor was evaluated for each propellant combination. These factors are listed in Table 22.

SOLID PROPELLANT COMBINATION COMPARISON

In this section, the solid propellant combinations listed in Table 10b are evaluated and compared. These comparisons contrast with those for the liquid propellants in that they are more general and are essentially a discussion of the capabilities of various solid propellant combinations. This approach was used because of the small number of solid propellants that were considered.

The areas in which the solid propellants were compared were equivalent to those considered for the liquid propellant combinations. These areas were performance, stop and restart capability, space storage, propellant hazards, and system design considerations. As mentioned previously, the solid propellants were placed in Phase II for a 1975 operational date. All discussion was based on this operational date.

PROPELLANT PERFORMANCE

Figure 10 summarizes the data in Table 10b schematically. Contours of the individual areas have been selected somewhat arbitrarily. Wide ranges of impulse and density are available within a given propellant family. However, only certain ranges are of practical interest. The large area shown for beryllium hydride propellants reflects the lack of specific information. This figure illustrates the effect of the above theoretical comments, namely, that high specific impulse almost inevitably results in lower density. This results from the fact that the formulation includes a light metal, a source of oxygen or fluorine and as much hydrogen as possible in the form of some solid derivative. A corollary of this conclusion is also evident, namely, that high impulse with "higher" density implies higher temperature because higher density means less hydrogen and higher average molecular weight. Approximate temperature isotherms have been sketched on Fig. 10 to illustrate this point. These should not be taken too literally. They do illustrate the point that solid propellants with theoretical specific impulse greater than 300 seconds are expected either to contain beryllium (as its hydride) or to have chamber temperatures in the 3500-4500 K range, (or both). The zero payload contours of Fig. 2 were superimposed on Fig. 10. It can be seen only a few combinations provide increases in payload over the N_2O_4/50-50 propellant combination.

STOP-RESTART AND THROTTLING

Although a flight weight, high performance stop-restart solid propulsion unit has not yet been made, this capability may be feasible

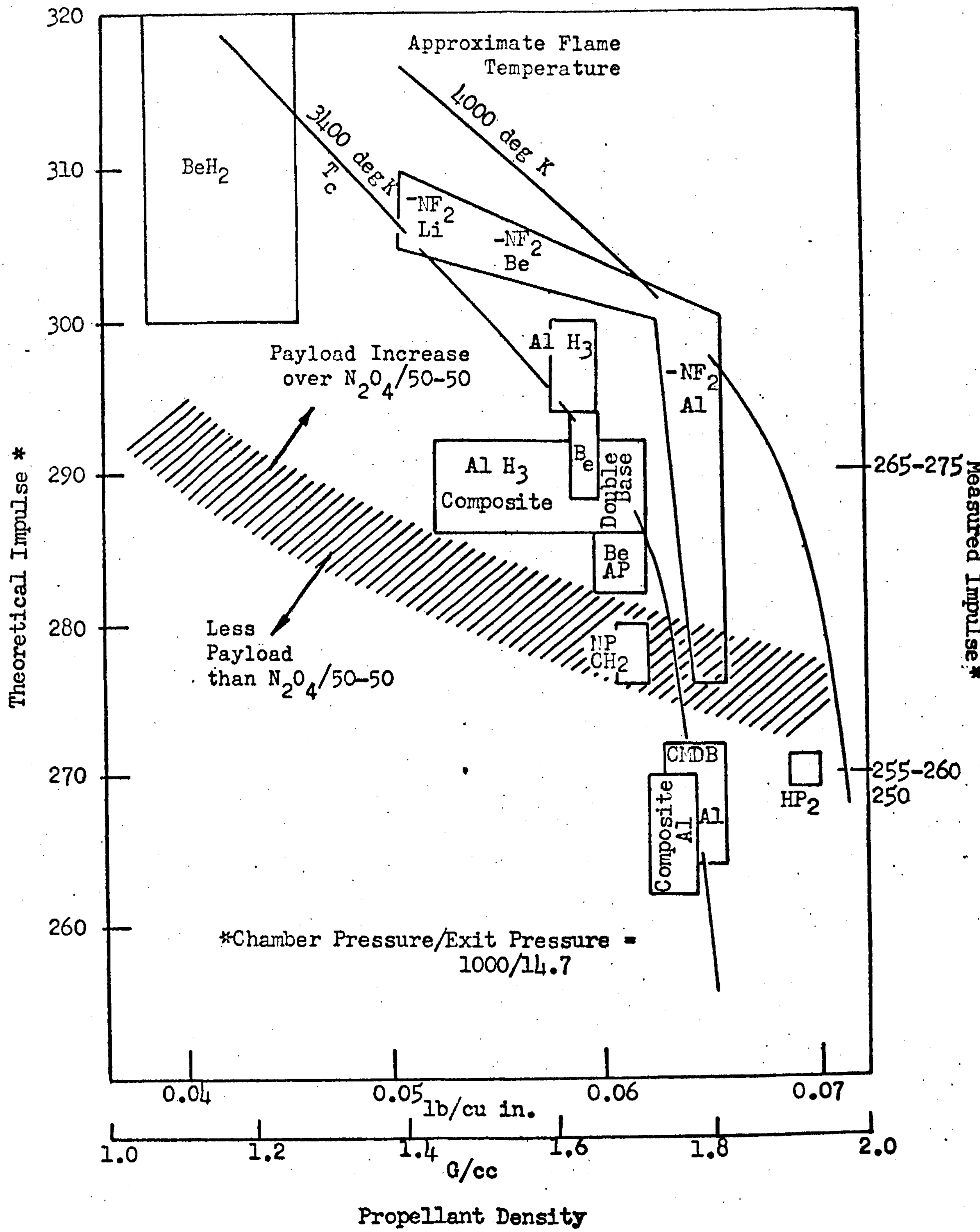

Fig. 10. Solid Propellant Families Impulse and Density Relations

in the 1970-1975 period. Two basic approaches have been taken to this propulsion capability. The first is a solid-pulse motor in which present propellant increments are fired on command. The second is based on burning rate control, including quenching, by control of chamber pressure and pressure gradients, (Ref. 5).

The pulse motor concept is a fairly straightforward engineering development which in its simplest form requires no moving parts. Problem areas associated with it are first the apparent intrinsic propellant fraction penalty associated with the restrictor or inhibitor separating the individual propellant increments and, second, the increased complexity of the ignition system which now is based either on a conductive film ignitor or multiple pyrotechnic charges. Alternatively, hypergolic ignition with chlorine trifluoride might be applicable if the restrictor can be removed or if it is feasible to burn it off with the hypergol.

The concept of quenching a solid propellant by a sudden pressure drop has been studied for various reasons for at least six years. During this period, the feasibility has been demonstrated for various modified double-base propellants and composite ammonium perchlorate or mixed oxidizer propellants with and without aluminum. Without going into detail, the data show that quenching is related to interruption of the heat flux from the flame into the propellant. Pressure decay rate required for quenching then depends on propellant composition and physical properties. The important chemical and physical features broadly are those which control the autoignition temperature under actual motor conditions. Chemically, the two most important factors are the decomposition kinetics of the oxidizer and fuel. The kinetically-controlling fuel is usually the binder rather than a metal. These kinetics may also depend on physical characteristics of the propellant, i.e., particle size of the oxidizer. In a sense, the solid-hybrid throttleable motor (Ref. 5) represents an extreme case of this physical separation. In this motor there are two grains, a fuel-rich one and an oxidizer-rich afterburner with a valve between. The fuel-rich grain is easily quenched because of its stoichiometry.

The investigation described above indicate that development of a stop-restart capability for solid propellant systems may be possible although a specific system cannot be defined at the present. Throttling is in one sense a special, much easier case of stop-restart. It has already been demonstrated in principle and in flight-weight prototype (Ref. 6) using nozzles with centerbodies (pintles or spikes) which permit change in throat area. There are engineering problems associated with the hardware design, but the principle is well established.

STABILITY IN SPACE

In designing a solid propellant motor with stop-restart capability for use in space, it was assumed that after the initial start-stop cycle, the propellant remaining in the motor will be exposed to the vacuum characteristic of its space environment. It is conceivable that nozzle closures could be devised, but conservative design would require that we assume the most stringent environmental conditions will prevail. Similarly, the motor will be exposed to whatever radiation is passing through the area. The effect of radiation levels ordinarily encountered in space upon solid propellant motors has been debated in a number of studies with varying conclusions. In general, the effect of radiation upon propellant characteristics is considered to be of low or secondary importance. During short missions of the order of a few days or weeks, the total dose normally is estimated to be below the threshold at which observable damage occurs. A large amount of protection is given to the propellant by even a relatively thin case.

Based on these assumptions, the greatest environmental stress on the propellant will result from exposure to high vacuum and the temperature-cycling characteristic of that portion of the vehicle. State-of-the-art solid propellants have been exposed for extended periods to vacuum, temperature cycling and high energy radiation. Observable changes in propellant characteristics have resulted from these exposures. The data show, as might be expected, that certain types of solid propellant binders are more desirable than others. However, none of the data indicate that solid propellants cannot be used in space missions. Some oxidizer and fuel ingredients now being considered for more advanced propellants may not be suitable for extended operations in space because of relatively high vapor pressure. For example, hydrazine nitroform has a relatively high sublimation pressure and formulations based on this may not be suitable for extended storage in high vacuum.

PROPELLANT HAZARDS

The two most important hazards associated with solid rocket propellants are toxicity and explosion. It is evident from the analysis presented above that propellants with theoretical specific impulse greater than 300 seconds, will probably contain beryllium. The only possible exceptions are very hot lithium-fluorine systems and possibly some of the aluminum hydride formulations with very energetic oxidizers.

Stability, compatibility and perhaps combustion problems can be anticipated for these possible exceptions. The possible hazards of using propellants containing beryllium therefore should be considered as perhaps a necessary adjunct to any practical solid propellant with very high theoretical impulses.

Propellants containing beryllium are now being developed routinely. There is no insurmountable problem associated with normal manufacture and use. There is a certain amount of inconvenience associated with the handling and test of these materials, but these problems have been solved by normal industrial hygiene precautions. The only serious problem is the occurrence or an incident releasing airborne beryllium containing material to which unprotected personnel could be exposed. The only serious hazard arises from inhalation of airborne material.

Accepted maximum allowable concentrations are 25 micrograms per cubic meter for a single exposure or 2 micrograms per cubic meter for an 8-hour day. These limits are for personnel without respiratory protection. Relatively simple respiratory protection, if used properly, can provide adequate protection against much higher concentrations. The acute exposure limits have also been expressed in terms of a total integrated dose of 500 microgram minutes per cubic meter. Based on these limits, the risk of exposing unprotected personnel to concentrations exceeding approved limits may be estimated from quantity-distance-meterological data.

Aluminum hydride is a possible nontoxic alternative to beryllium metal, i.e., reaching a similar impulse range. However, the stability of aluminum hydride is marginal. A fairly radical improvement in stability is needed before it would be suitable for formulations with very high reliability.

Other materials to be used in advanced solid propellants will have toxic properties. Fluorine is probably the ingredient next most likely to cause concern. Hazards associated with hydrogen fluoride which would be the most likely combustion product are well known. They differ from beryllium in that complete recovery from acute exposure can perhaps be anticipated.

Explosion hazards associated with solid propellants are well known and need not be discussed in detail. In most instances, propellants present primarily a fire hazard. A major distinction of interest where large quantities of propellant are involved is the ability of the propellant to propagate detonation waves. Generally speaking, solid propellants which contain as a continuous phase an ingredient such as nitrocellulose, which is itself a propellant or explosive, can propagate a detonation wave. If the continuous phase does not have monopropellant characteristics, the composition will not propagate a detonation except in very large masses. The composite propellants based on predominantly hydrocarbon binders are good examples of the class which does not detonate.

MOTOR-DESIGN CONSIDERATIONS

Other factors besides those discussed above would also require consideration in selecting a propellant for a specific motor design. For example, range of burning rate available affects grain design. At present, a range of burning rates from about 0.1 in/sec to 5 or 10 in/sec at 1000 psia can be envisioned. There is no guarantee that propellants with this range of burning rates will necessarily have all of the other characteristics such as stop-restart capability, but it does illustrate the fact that a burning-rate range of two orders of magnitude can be predicted. Very high burning rates (5-10 in/sec) will probably not be true propellant regression rates but will depend on devices to increase the mass consumption rate and hence the effective "burning rate."

Physical properties of these advanced propellants may also be expected to vary widely from the conventional viscoelastic composite solid to the nearly rigid reinforced structures based on inclusion of wire, (Ref.7) screen or other structural member in the propellant. The latter structures also have anisotropic burning rates, thus providing an additional degree of freedom in grain design (Ref. 8).

Similarly, various forms of solid - solid hybrid may be predicted, especially for very high impulse propellants based on beryllium hydride for which the large volume fraction of fuel makes conventional formulation impractical. This will impose some limitations on motor design not as yet defined. However, for example, one would predict that as large an L* as possible should be used.

Techniques for ignition in vacuum will be available. Multiple pyrogens are preferred now for reliability. Hypergolic materials such as chlorine, trifluoride or fluorine have attractive features if their performance in vacuum can be improved. If a liquid hypergol is used, this provides the further possibility of using the hypergol not only for ignition but also for thrust augmentation during mainstage operation, i.e., a simple form of hybrid.

Propellants based on $-NF_2$ or hypothatical solid -OF oxidizers have within the limits of present knowledge several undesirable features which may be inherent. First, all formulations made thus far have been shock sensitive. Second, high-impulse formulations (as yet hypothetical) have high combustion temperatures. However, lithium-based formulations in theory could provide nontoxic propellants with theoretical impulse greater than 300 seconds.

SOLID PROPELLANT SELECTION

Based on previous discussion, solid propellant choices for the 1975 lunar mission would be either:

(1) A beryllium-hydride composite probably based on a conventional oxidizer such as ammonium perchlorate and a solid - solid hybrid formulation technique

or:

(2) A beryllium formulation using an oxygen-based oxidizer with theoretical impulse in the 290-295 second range, perhaps using reinforcing techniques to provide maximum strength and resistance to temperature cycling.

These selections were based primarily on high performance and good stability characteristics. The solid propellant combinations selected (particularly the first combination) could provide better performance than the N_2O_4/50-50 systems. However, the stop-restart capability has not been sufficiently developed for the solid propellants to merit their further, more-detailed consideration as propellants for the Apollo systems.

PROPELLANT SELECTIONS

Following their development and evaluation, the propellant combination comparison factors were grouped into five basic comparison areas: (1) Performance, (2) Reliability, (3) Operation Aspects, (4) Development Ease, and (5) Launch Operation Ease. The comparison factors were weighted and combined such that a rating of 100 was possible in each of the five areas. Each of the five basic comparison areas was then weighted according to relative importance to provide an overall, numerical evaluation criterion.

An evaluation scheme of this numerical nature has two significant features:

1. The evaluation and comparison of propellants is systematic, and
2. The importance of each factor contributing to a given rating can immediately be determined and isolated for review.

Obviously the comparison factor combinations were different for the 1970 and 1975 operational dates. These factor combinations and relative weighting are presented in Tables 34 and 35 .

For the 1970 operational date, it was assumed that the performance, reliability and operational aspects, and the development and launch ease should be approximately equal in importance. The weightings were assigned as shown in Table 34 with the launch ease of considerably less importance than the other areas. With these definitions, the overall propellant combination evaluation factor can be determined. The maximum possible value of this factor was 300.

In weighting the areas for the 1975 date, considerably greater emphasis was placed upon performance and less emphasis was placed upon development ease. The assigned weighting are shown in Table 35 . A maximum overall rating of 300 was also possible for this 1975 evaluation.

An example of the combination of the comparison factors into the overall rating is provided in Table 36 for two of the propellant candidates. The individual comparison factors are listed, the five evaluation-area factors evaluated, and then the overall rating factor listed. This illustration is based upon the 1970 operation date.

TABLE 34

1970 PROPELLANT COMBINATION RATING

Main Rating Area	Weighting Factor 1970	Rating Areas	Weighting Factor
I. Performance	1.0	A. Relative Payload	8
		B. Relative Volume	2
II. Reliability	0.5	A. Experience	4
		B. Propellant Transfer Method	2
		C. Operation Simplicity	4
III. Operational Aspects	0.5	A. Operational Simplicity	3
		B. Operational Sensitivity	1.5
		C. Propellant Thermal Storage	4.0
		D. Thrust Chamber Cooling	1.5
IV. Development Ease	0.8	A. Experience	2
		B. Relative Volume	1.5
		C. Toxicity	1
		D. Availability	2
		E. Propellant Transfer Method	1
		F. Thrust Chamber Cooling	2.5
V. Launch Operation Ease	0.2	A. Launch Storage Method	5
		B. Toxicity	5

TABLE 35
1975 PROPELLANT COMBINATION RATINGS

Main Rating Area	Weighting Factor	Rating Areas	Weighting Factor
I. Performance	1.8	A. Relative Payload	10
II. Reliability	0.25	A. Propellant Transfer	3.5
		B. Method Operation Simplicity	6.5
III. Operational Aspects	0.6	A. Operation Simplicity	3
		B. Operation Sensitivity	1.5
		C. Propellant Thermal Storage	4.0
		D. Thrust Chamber Cooling	1.5
IV. Development Ease	0.20	A. Thrust Chamber Cooling	6
		B. Toxicity	2
		C. Propellant Transfer Method	2
V. Launch Operation Ease	0.15	A. Launch Storage Method	5
		B. Toxicity	5

Table 36

PROPELLANT COMBINATION RATING ILLUSTRATION
1970 Operational Data

Propellants	F_2/N_2H_4	OF_2/B_2H_6
I. Comparison Factors		
A. Relative Payload	8.8	7.7
B. Relative Volume	9.0	7.2
C. Experience	6.5	6.0
D. Propellant Transfer Method	6.0	6.0
E. System Simplicity	7.0	7.0
F. System Sensitivity	7.7	8.1
G. Logistics	9.0	6.0
H. Space Storage	7.9	7.4
I. Thrust Chamber Cooling	6.0	2.0
J. Launch Storage	6.0	4.0
K. Toxicity	5.0	0.0
II. Evaluation Areas		
A. Performance 8 (IA) + 2 (IB)	88.4	76.0
B. Reliability	66.0	64.0
C. Operational Aspects	73.1	65.8
D. Development Ease	70.5	45.1
E. Launch Operation Ease	55.0	20.1
III. Overall Rating	225	181

1970 PROPELLANT-COMBINATION COMPARISON

Based upon the method described for 1970 in Table 34, the five evaluation-area factors and the overall comparison factor were determined. These factors are listed for various propellant combinations in Tables 37 through 42. No hydrogen-fueled propellant combinations appear in the tables. They were excluded from the 1970 listings because of their low density, and the restriction that major (tank) modification be considered only for the 1975 systems. Also, only propellant combinations offering a performance improvement over N_2O_4/50-50 are listed.

The performance ratings (basically the payload gain comparison) are predominantly oxidizer-oriented. The higher ratings are achieved by the F_2, FLOX(90), and OF_2 oxidizers with moderately dense fuels. Oxidizers with lower ratings are the N_2F_4, NF_3, and Comp. A.

In Table 38, the reliability ratings are listed. These ratings are predominantly experience-oriented. (Only propellants which offer performance improvements over N_2O_4/50-50 are being considered, thus many propellants, for which there is extensive development experience, have already been eliminated.) The N_2O_4/50-50 combination is included only as a reference. As might be expected, it ranks as the highest combination in this and the following areas.

In the operational-aspects area, Table 39, the higher-ranking propellant combinations are the noncryogenic, "earth storable". The highest-ranking cryogenic is the FLOX(90) followed by N_2F_4 and OF_2. Propellant storage and simplicity are the predominant influences. The development ratings, listed in Table 40, are experience-oriented and modified by the thrust chamber cooling comparison. The effect of the cooling factor can be seen by the relatively low ranking of several of the B_5H_9 combinations.

The launch operation ratings are presented in Table 41. The two factors involved are toxicity and launch storage ease. Effects of toxicity can be noted by the low ranking of the OF_2 and B_5H_9 combinations. The highest rankings are achieved by the noncryogenic and/or nontoxic propellants.

TABLE 37

1970 PROPELLANT COMBINATION PERFORMANCE RATING

Propellant Combination	Rating	Propellant Combination	Rating
F_2/N_2H_4	88.4	OF_2/CH_4	68.4
F_2/B_2H_6	87.6	OF_2/N_2H_4	67.6
F_2/B_5H_9	84	N_2F_4/B_5H_9	66.8
FLOX(90)/50-50	83.6	N_2F_4/N_2H_4	66.8
FLOX(90)/MMH	82.2	OF_2/UDMH	66.4
F_2/NH_3	81	FLOX(90)/C_2H_6	63.6
$OF_2/C_2H_5B_{10}H_{13}$	78.4	F_2/Hydyne	62.6
OF_2/B_5H_9	77.6	F_2/CH_4	60.4
FLOX(90)/UDMH	77.0	OF_2/C_2H_6	60.0
F_2/50-50	76.8	OF_2/NH_3	54.0
OF_2/B_2H_6	76.0	FLOX(30)/B_2H_6	53.6
F_2/MMH	73.2	NF_3/N_2H_4	51.4
OF_2/MMH	71	N_2F_4/NH_3	51.4
OF_2/50-50	70.2	FLOX(30)/B_5H_9	50.6
FLOX(90)/CH_4	70	Comp A/N_2H_4	48.8
OF_2/RP-1	69.6	Comp A/Hybaline A-5	45.6
F_2/UDMH	69.0	FLOX(30)/N_2H_4	41.6

TABLE 38

1970 PROPELLANT COMBINATION RELIABILITY RATING

Propellant Combination	Rating	Propellant Combination	Rating	Propellant Combination	Rating
N_2O_4/50-50	88	FLOX(30)/NH_3	56	FLOX(90)/C_2H_6	48
O_2/RP-1	72	Comp A/N_2H_4	54	Comp A/50-50	48
FLOX(90)/RP-1	72	FLOX(90)/UDMH	54	ClF_3/Hybaline A-5	46
N_2O_4/N_2H_4	70	FLOX(90)/MMH	54	N_2O_4/B_5H_9	46
F_2/NH_3	68	FLOX(30)/N_2H_4	54	O_2/B_5H_9	46
ClF_3/50-50	68	O_2/UDMH	54	Comp A/MMH	46
N_2H_4/B_5H_9	68	OF_2/UDMH	54	F_2/C_2H_6	45
MON/MMH	66	OF_2/N_2H_4	54	ClF_3/Hydrazoid-P	44
ClF_3/N_2H_4	66	OF_2/MMH	54	$OF_2/C_2H_5B_{10}H_{13}$	44
F_2/N_2H_4	66	OF_2/B_5H_9	54	N_2O_4/B_2H_6	44
F_2/MMH	66	O_2/CH_4	52	O_2/B_2H_6	44
N_2O_4/Hybaline A-5	66	F_2/RP-1	52	H_2O_2/B_2H_6	44
OF_2/B_2H_6	64	F_2/50-50	52	NF_3/B_5H_9	44
F_2/B_2H_6	64	OF_2/NH_3	52	NF_3/N_2H_4	44
O_2/N_2H_4	64	OF_2/RP-1	52	MON/B_5H_9	42
O_2/50-50	64	F_2/UDMH	52	N_2F_4/NH_3	42
H_2O_2/B_5H_9	62	FLOX(30)/B_2H_6	52	N_2F_4/RP-1	
FLOX(90)/CH_4	62	ClF_3/MMH	50	Comp A/Hybaline A-5	42
OF_2/CH_4	58	F_2/B_5H_9	50	NF_3/UDMH	40
ClF_3/B_5H_9	58	FLOX(30)/B_5H_9	50	OF_2/H_2	40
FLOX(90)/50-50	56	F_2/CH_4	48	N_2F_4/MMH	40
OF_2/50-50	56	N_2F_4/N_2H_4	48	N_2F_4/UDMH	40
F_2/H_2	56	N_2F_4/B_5H_9	48		

TABLE 39

1970 PROPELLANT COMBINATION OPERATIONAL ASPECTS RATING

Propellant Combination	Rating	Propellant Combination	Rating	Propellant Combination	Rating
N_2O_4/50-50	88.2	N_2F_4/N_2H_4	75.7	F_2/B_2H_6	68.2
ClF_3/Hydrazoid-P	86.6	FLOX(90)/C_2H_6	75.6	$OF_2/C_2H_5B_{10}H_{13}$	68.1
ClF_3/MMH	86.6	F_2/RP-1	74.7	OF_2/B_5H_9	68.1
Comp A/MMH	85.8	FLOX(90)/UDMH	74.6	FLOX(30)/B_2H_6	67.6
ClF_3/50-50	85.4	F_2/NH_3	74.3	OF_2/C_2H_6	67.6
ClF_3/N_2H_4	85.4	FLOX(90)/CH_4	73.9	O_2/B_5H_9	67.1
Comp A/50-50	85.0	F_2/50-50	73.2	N_2F_4/B_2H_6	67.1
Comp A/N_2H_4	84.3	F_2/N_2H_4	73.1	F_2/B_5H_9	67.0
N_2H_4/B_5H_9	83.2	F_2/MMH	73.1	OF_2/Hydrazoid-P	66.6
MON/MMH	82.0	OF_2/N_2H_4	72.6	H_2O_2/B_2H_6	66.6
N_2O_4/MMH	81.8	N_2F_4/B_5H_9	71.9	OF_2/CH_4	66.1
ClF_3/Hybaline A-5	81.7	N_2O_4/B_5H_9	71.7	O_2/N_2H_4	66.0
Comp A/Hybaline A-5	81.3	OF_2/UDMH	71.6	OF_2/B_2H_6	65.8
ClF_3/B_5H_9	81.3	OF_2/MMH	71.6	O_2/RP-1	64.0
FLOX(90)/MMH	80.9	FLOX(30)/B_5H_9	71.5	O_2/50-50	64.0
FLOX(30)/NH_3	80.4	FLOX(90)/50-50	71.5	NF_3/B_5H_9	64.0
N_2F_4/50-50	78.8	N_2F_4/CH_4	71.4	N_2O_4/B_2H_6	63.6
N_2F_4/NH_3	78.6	OF_2/50-50	71.2	O_2/CH_4	63.5
N_2F_4/UDMH	78.4	H_2O_2/B_5H_9	71.1	O_2/B_2H_6	58.7
OF_2/NH_3	78.3	F_2/CH_4	70.3	F_2/H_2	33.2
N_2F_4/RP-1	77.9	MON/B_5H_9	70.2	NF_3/H_2	27.6
F_2/UDMH	76.6	NF_3/N_2H_4	68.9	OF_2/H_2	26.4
FLOX(30)/N_2H_4	75.9	NF_3/UDMH	68.9		

TABLE 40

1970 PROPELLANT COMBINATION DEVELOPMENT RATING

Propellant Combination	Rating	Propellant Combination	Rating
N_2O_4/50-50	78.0	ClF_3/Hydrazoid-P	59.4
O_2/RP-1	75.7	FLOX(90)/UDMH	59.3
F_2/NH_3	75.3	N_2F_4/N_2H_4	58.6
N_2O_4/N_2H_4	73.3	Comp A/N_2H_4	58.3
ClF_3/50-50	71.1	F_2/B_2H_6	58.3
F_2/N_2H_4	70.5	FLOX(90)/C_2H_6	58.2
F_2/MMH	70.4	FLOX(90)/50-50	57.9
ClF_3/N_2H_4	70.2	Comp A/50-50	57.3
O_2/N_2H_4	69.7	Comp A/MMH	56.4
FLOX(30)/NH_3	68.9	N_2F_4/CH_4	56.4
MON/MMH	68.2	N_2F_4/RP-1	56.3
FLOX(90)/MMH	68.2	OF_2/NH_3	55.6
F_2/UDMH	67.8	NF_3/N_2H_4	55.3
O_2/50-50	66.9	N_2F_4/UDMH	55.2
F_2/RP-1	66.8	N_2F_4/50-50	55.0
FLOX(90)/CH_4	65.2	H_2O_2/B_5H_9	54.0
F_2/H_2	65.0	OF_2/RP-1	53.1
O_2/CH_4	64.2	NF_3/UDMH	52.7
N_2H_4/B_5H_9	63.6	ClF_3/B_5H_9	51.8
F_2/50-50	63.2	OF_2/N_2H_4	50.4
FLOX(30)/N_2H_4	62.3	ClF_3/Hybaline A-5	50.3
O_2/UDMH	62.2	OF_2/C_2H_6	49.5
ClF_3/MMH	62	OF_2/50-50	49.1
N_2F_4/NH_3	61.1	OF_2/MMH	48.1
OF_2/CH_4	60.2	OF_2/H_2	48.0
F_2/CH_4	59.6		

TABLE 41

1970 PROPELLANT COMBINATION LAUNCH OPERATION RATING

Propellant Combination	Rating	Propellant Combination	Rating	Propellant Combination	Rating
N_2O_4/Hybaline A-5	90	N_2F_4/CH_4	60	N_2F_4/B_2H_6	40
O_2/RP	90	N_2O_4/B_2H_6	60	F_2/B_2H_6	40
N_2O_4/N_2H_4	85	H_2O_2/B_2H_6	60	O_2/B_5H_9	40
N_2O_4/50-50	80	Comp A/N_2H_4	60	FLOX(30)/B_2H_6	40
O_2/CH_4	80	Comp A/MMH	60	FLOX(90)/C_2H_6	35
O_2/N_2H_4	75	Comp A/Hybaline A-5	60	F_2/C_2H_6	35
O_2/UDMH	75	Comp A/50-50	60	$OF_2/C_2H_5B_{10}H_{13}$	30
O_2/50-50	70	F_2/N_2H_4	55	NF_3/B_5H_9	30
N_2F_4/RP-1	70	F_2/MMH	55	FLOX(30)/B_5H_9	30
FLOX(90)/MMH	70	F_2/50-50	55	N_2F_4/B_5H_9	30
NF_3/H_2	70	F_2/UDMH	55	MON/B_5H_9	30
MON/MMH	65	F_2/RP-1	55	F_2/B_5H_9	30
N_2F_4/N_2H_4	65	FLOX(30)/N_2H_4	55	ClF_3/B_5H_9	30
N_2F_4/MMH	65	FLOX(90)/50-50	55	OF_2/N_2H_4	30
N_2F_4/UDMH	65	FLOX(90)/UDMH	55	OF_2/MMH	30
NF_3/UDMH	65	FLOX(90)/RP-1	55	OF_2/50-50	30
NF_3/N_2H_4	65	N_2H_4/B_5H_9	50	OF_2/UDMH	30
ClF_3/N_2H_4	60	N_2O_4/B_5H_9	50	OF_2/RP-1	30
ClF_3/MMH	60	O_2/B_2H_6	50	OF_2/B_5H_9	30
ClF_3/50-50	60	H_2O_2/B_5H_9	50	OF_2/NH_3	20
ClF_3/Hydrazoid-P	60	F_2/NH_3	45	OF_2/CH_4	20
ClF_3/Hybaline A-5	60	F_2/CH_4	45	OF_2/B_2H_6	20
N_2F_4/NH_3	60	FLOX(30)/NH_3	45	OF_2/H_2	20
N_2F_4/50-50	60	FLOX(90)/CH_4	45	OF_2/C_2H_6	10

TABLE 42

1970 PROPELLANT COMBINATION OVERALL RATING

Propellant Combination	Rating	Propellant Combination	Rating	Propellant Combination	Rating
F_2/N_2H_4	225	OF_2/N_2H_4	177	Comp A/50-50	162
F_2/NH_3	221	OF_2/RP-1	177	O_2/RP-1	162
FLOX(90)/MMH	218	Comp A/N_2H_4	177	ONF_3/B_5H_9	162
F_2/MMH	210	F_2/CH_4	176	N_2F_4/50-50	161
F_2/B_2H_6	208	F_2/RP-1	174	Comp A/MMH	159
FLOX(90)/50-50	206	OF_2/UDMH	173	ONF_3/N_2H_4	158
F_2/50-50	201	ClF_3/N_2H_4	172	O_2/50-50	158
FLOX(90)/UDMH	200	N_2F_4/NH_3	172	Comp A/Hybaline A-5	157
F_2/UDMH	199	N_2O_4/N_2H_4	172	MOXIE 2A/NH_3	156
FLOX(90)/CH_4	199	N_2H_4/B_5H_9	171	FLOX(30)/B_2H_6	155
F_2/Hydrazoid-P	190	OF_2/NH_3	171	ClF_3/MMH	155
N_2F_4/N_2H_4	189	F_2/C_2H_6	171	FLOX(30)/NH_3	154
F_2/B_5H_9	187	$OF_2/C_2H_5B_{10}H_{13}$	170	N_2F_4/CH_4	154
N_2O_4/50-50	183	N_2F_4/B_5H_9	168	MOXIE 2A/Hydrazoid-P	154
OF_2/CH_4	182	FLOX(30)/N_2H_4	167	F_2/H_2	153
OF_2/B_2H_6	181	N_2F_4/MMH	166	ClF_3/Hydrazoid-P	153
OF_2/50-50	179	N_2F_4/50-50	165	FLOX(30)/B_5H_9	153
FLOX(90)/C_2H_6	179	NF_3/N_2H_4	165	OF_2/C_2H_6	153
OF_2/B_5H_9	179	ClF_3/50-50	165	MOXIE 2A/N_2H_4	152
OF_2/MMH	178	O_2/N_2H_4	164	MOXIE 2A/MMH	150

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The overall ratings for propellant combinations for 1970 are listed in Table 42 . The F_2/N_2H_4 ranks the highest and the F_2 oxidizer combinations in general occupy the highest ranking positions (FLOX (90) is 90 percent F_2 and 10 percent O_2). The N_2F_4/N_2H_4 propellant combination is the next oxidizer in rank and is followed by the OF_2-oxidizer combinations. All of the top-ranking combinations are at least partially cryogenic (i.e., the oxidizer is a cryogenic). The top-ranking combination that is completely noncryogenic or "earth storable" is Comp A/N_2H_4. This is followed closely by ClF_3/N_2H_4, N_2O_4/N_2H_4, and N_2H_4/B_5H_9. (As mentioned previously, the N_2O_4/50-50 combination is included as a reference to illustrate its high quality although it does not compete with the other propellants on the basis of payload.)

PROPELLANT-CANDIDATE SELECTION - 1970

From this propellant-combination rating, four candidate combinations were selected for the more detailed evaluations of Task II. To enable the Task II investigation to provide a distinctive propellant comparison with a broad scope of propulsion-system configurations, candidate propellant combinations having different characteristics were selected. This selection will ensure that should undesirable features of a given propellant (oxidizer or fuel) be uncovered, all of the candidates will not be affected and the analyses can proceed without interruption. Four high ranking oxidizers were chosen: F_2, FLOX(90), OF_2, and Comp A. The Comp A oxidizer was included as the highest ranking noncryogenic "earth storable" oxidizer. Fuels which give the best rankings were then selected for each of these oxidizers. The selections are given in Table 43 . Multiple fuels are indicated for the F_2 and OF_2. These fuels are all high ranking and provide some flexibility in the thrust chamber cooling analyses. A single fuel will be selected after more detailed consideration.

TABLE 43

1970 PROPELLANT COMBINATION CANDIDATES

F_2/N_2H_4; NH_3

OF_2/CH_4; B_2H_6; MMH;

FLOX(90)/MMH

Comp A/N_2H_4

A list of the properties of these combinations is given in Table 44 .

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TABLE 44

1970 PROPELLANT COMBINATION COMPARISON

Propellant Combination	Weight Mixture Ratio	Volume Mixture Ratio	Theoretical Performance: Sea Level** Specific Impulse	Theoretical Performance: Vacuum*** Specific Impulse	Bulk Specific Gravity	Hyperbolic	Combustion Temperature F
F_2/N_2H_4	2.3	1.51	363	422	1.31	Yes	7550
$F_2/N_2H_4 + C_{10}H_{16}O_8N_2$*	2.3	1.51	362	420	1.31	Yes	7500
F_2/NH_3	3.3	1.46	359	419	1.18	Yes	7370
FLOX(90-10)/MMH	2.7	1.61	356	416	1.23	Yes	6640
OF_2/CH_4	5.6	1.55	348	408	1.09	No	6960
OF_2/MMH	2.5	1.45	343	403	1.26	Yes	6850
OF_2/B_2H_6	3.6	1.04	365	433	0.99	Yes	7440
Comp. A/N_2H_4	2.7	1.42	313	361	1.47	Yes	6620
Comp. A/N_2H_4 + $C_{10}H_{16}O_8N_2$	2.7	1.42	312	360	1.47	Yes	6590

* 1 percent additive to stabilize N_2H_4

** Chamber Pressure, psia = 1000,
Expansion Area Ratio = Optimum Sea Level

*** Chamber Pressure, psia = 300
Expansion Area Ratio = 40

1975 PROPELLANT-COMBINATION COMPARISON

In the selection of propellant-combination candidates for the 1975 operational dates, there were two objectives. First, the 1975 propellant-combination candidates must provide a payload capability comparable to the 1970 propellant candidates. Second, it was desired that representatives of the various propellant physical states be included regardless of their position in the overall ranking.

With the first objective in mind, only propellant combinations with a relative payload capability factor greater than 5.5 were considered. This limiting value was established from a consideration of the 1970 propellant-combination candidates. To facilitate the second objective, the propellant combinations were evaluated and grouped according to their physical state: liquids, hybrids, solids, and solid additive. Although comparisons can be made from group to groups, emphasis was on the intergroup comparison.

In making selections for the 1975 candidates, only the liquids, hybrids, and solid additive propellants were considered. Although the solid propellants listed could be of interest, there is the area of solid propellant start and cutoff technology that must be developed before they can be considered. For space missions, the start and cutoff capability is extremely important and, although this capability may be developed by 1975, investigations in this area were considered outside the scope of the contract. Therefore, of the six candidates selected, three were bipropellant liquids, one was a hybrid, and two use solid additives to a liquid bipropellant system.

The bipropellant combinations for 1975 were rated within the five evaluation areas of Table 35 . In Table 45 are presented the performance ratings of the bipropellant combinations considered for 1975. Unlike the similar ratings for 1970, these ratings do not include the effect of relative volume. The 1975 performance rating is a function of only the payload. For this reason, propellant combinations with hydrogen as a fuel rank among the top candidates. The payloads for the hydrogen-fueled combinations were calculated using a light tank-weight factor as described previously.

All of the combinations have fluorine-based oxidizers with the exception of the one system, O_2/H_2. This combination ranks high by virtue of the hydrogen fuel. Since the 1975 propellant selection

TABLE 45

1975 BI-PROPELLANT COMBINATION PERFORMANCE RATING

Propellant Combination	Rating	Propellant Combination	Rating
		OF_2/CH_4	66
F_2/H_2	116	OF_2/MMH	66
OF_2/H_2	94	OF_2/RP-1	66
F_2/B_2H_6	89	OF_2/50-50	66
F_2/N_2H_4	88	F_2/UDMH	65
F_2/B_5H_9	84	N_2F_4/B_2H_6	64
FLOX(90-10)/50-50	83	ONF_3/B_5H_9	63
F_2/Hydrazoid-P	83	N_2F_4/N_2H_4	62
N_2F_4/H_2	82	OF_2/N_2H_4	62
F_2/NH_3	80	OF_2/UDMH	62
FLOX(90-10)/MMH	80	OF_2/Hydyne	60
$OF_2/C_2H_5B_{10}H_{13}$	79	FLOX(90-10)/C_2H_6	60
OF_2/B_2H_6	77	N_2F_4/B_5H_9	60
OF_2/B_5H_9	76	O_2/H_2	58
F_2/50-50	74	F_2/Hydyne	58
FLOX(90-10)/UDHM	72	OF_2/Hydrazoid-P	58
F_2/MMH	70	F_2/CH_4	57
FLOX(90-10)/CH_4	68	FLOX(30-70)/B_2H_6	56

is heavily weighted in favor of performance, the status of the propellant combination in this rated area (Table 45) is fairly indicative of relative overall ranking.

The reliability ratings of the propellant combinations are given in Table 46. A rating in this area reflects the two factors which establish reliability of the propulsion system. These two factors are: (1) propellant transfer method and (2) operation simplicity. Experience, a third factor considered in determining a 1970 reliability rating, is a less important parameter in rating for 1975. The time span is adequate to gain sufficient experience with any of the propellant combinations for which experience is lacking.

As seen in Table 46 , the ratings within the area are very insensitive to the propellant combination. Since each combination is a liquid bipropellant, the method of propellant transfer is similar and there is differentiation of the propellant combinations. Therefore, operation simplicity is the differentiating feature. The more complex hydrogen-fueled systems receive the lowest ratings. These low ratings counter in part the high performance ratings of the hydrogen-fueled combinations.

With the exception of the hydrogen-fueled systems, the ratings for the area of operational aspects are clustered in the range of 60 to 80 (Table 47). Within this range, the propellant combinations are fairly evenly distributed. Unlike the ratings for reliability, each propellant combination has a distinct rating which establishes its relative standing.

The ratings reflecting the ease of development of a propulsion system are presented in Table 48 . Essentially two factors: (1) thrust chamber cooling and (2) toxicity are responsibile for this ranking of the propellant combinations. Oxygen is the only nontoxic oxidizer in contention. Hydrogen is an excellent coolant. As expected from these two favorable characteristics, the oxygen/hydrogen combination is the highest rated system. Noticeably this combination has a significantly better rating than the second best combination. At the other extreme, OF_2/B_5H_9 receives the lowest rating. Individually, the propellants are exceedingly toxic. Coupled with this disadvantage is the high combustion temperature and the poor coolant properties of the fuel.

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TABLE 46

1975 BI-PROPELLANT COMBINATION RELIABILITY RATING

Propellant Combination	Rating	Propellant Combination	Rating
F_2/B_2H_6	66.5	OF_2/50-50	66.5
F_2/B_5H_9	66.5	FLOX(90-10)/C_2H_4	66.5
F_2/CH_4	66.5	FLOX(90-10)/C_2H_6	66.5
F_2/MMH	66.5	FLOX(90-10)/MMH	66.5
F_2/NH_3	66.5	FLOX(90-10)/UDMH	66.5
F_2/N_2H_4	66.5	FLOX(90-10)/50-50	66.5
F_2/UDMH	66.5	FLOX(30-70)/B_2H_6	66.5
F_2/50-50	66.5	OF_2/CH_4	53.5
F_2/Hydyne	66.5	OF_2/RP-1	53.5
F_2/Hydrazoid-P	66.5	N_2F_4/B_5H_9	53.5
OF_2/B_2H_6	66.5	N_2F_4/N_2H_4	53.5
OF_2/B_5H_9	66.5	N_2F_4/B_2H_6	53.5
$OF_2/C_2H_5B_{10}H_{13}$	66.5	ONF_3/B_5H_9	53.5
OF_2/Hydyne	66.5	F_2/H_2	47.0
OF_2/Hydrazoid-P	66.5	N_2F_4/H_2	34.6
OF_2/MMH	66.5	OF_2/H_2	34.0
OF_2/N_2H_4	66.5	O_2/H_2	34.0
OF_2/UDMH	66.5		

TABLE 47

1975 BI-PROPELLANT COMBINATION OPERATIONAL ASPECTS RATING

Propellant Combination	Rating	Propellant Combination	Rating
FLOX(90-10)/MMH	77.7	ONF_3/B_5H_9	70.8
N_2F_4/N_2H_4	75.7	FLOX(90-10)/CH_4	70.7
F_2/NH_3	74.3	F_2/CH_4	70.3
F_2/Hydyne	73.6	F_2/B_5H_9	70.1
F_2/50-50	73.2	F_2/Hydrazoid-P	69.1
F_2/N_2H_4	73.2	F_2/UDMH	68.6
F_2/MMH	73.1	F_2/B_2H_6	68.2
OF_2/Hydyne	73.1	FLOX(90-10)/50-50	68.2
$OF_2/C_2H_5B_{10}H_{13}$	72.6	OF_2/B_5H_9	68.1
OF_2/N_2H_4	72.6	FLOX(30-70)/B_2H_6	67.1
FLOX(90-10)/C_2H_6	72.3	N_2F_4/B_2H_6	67.1
N_2F_4/B_5H_9	72.0	OF_2/RP-1	66.7
OF_2/Hydrazoid-P	71.9	OF_2/CH_4	66.0
OF_2/MMH	71.6	OF_2/B_2H_6	65.8
OF_2/UDMH	71.5	F_2/H_2	33.2
FLOX(90-10)/CH_4	71.3	O_2/H_2	27.1
OF_2/50-50	71.2	OF_2/H_2	26.4
		N_2F_4/H_2	26.0

TABLE 48

1975 BI-PROPELLANT COMBINATION DEVELOPMENT RATING

Propellant Combination	Rating	Propellant Combination	Rating
O_2/H_2	86	F_2/B_2H_6	44
N_2F_4/H_2	76	N_2F_4/B_2H_6	44
FLOX(90-10)/MMH	76	OF_2/Hydyne	42
F_2/H_2	70	OF_2/N_2H_4	42
F_2/UDMH	70	OF_2/RP-1	42
N_2F_4/N_2H_4	68	$OF_2/C_2H_5B_{10}H_{13}$	42
F_2/MMH	64	F_2/Hydrazoid-P	40
F_2/NH_3	64	FLOX(30-70)/B_2H_6	38
FLOX(90-10)/CH_4	64	F_2/B_5H_9	36
FLOX(90-10)/C_2H_6	64	OF_2/MMH	36
F_2/CH_4	64	OF_2/UDMH	36
OF_2/H_2	60	OF_2/50-50	36
F_2/N_2H_4	58	N_2F_4/B_5H_9	36
F_2/Hydyne	58	ONF_3/B_5H_9	36
F_2/50-50	58	OF_2/Hydrazoid-P	36
OF_2/CH_4	54	OF_2/B_2H_6	24
FLOX(90-10)/UDMH	52	OF_2/B_5H_9	24
FLOX(90-10)/50-50	46		

The propellant combinations have been rated (Table 49) according to the ease in the launch operation of a missile using the propellants. The oxygen/hydrogen system ranks as the best combination primarily because it is nontoxic.

The area ratings were combined according to the weighting factors of Table 35 to give the overall ratings presented in Table 50. Heading the list is fluorine/hydrogen with a rating of 268. This system has a large payload capability which contributes heavily to the final overall rating. Performance was assigned the largest weighting factor of the five evaluation areas. It is interesting and significant that F_2/N_2H_4 ranks as the second best bipropellant for 1975. This combination received top ranking in the selection of propellants for 1970. Fluorine-oxidized combinations dominate the top echelon of propellants. Two FLOX (90-10) systems are in the top combination echelon. $OF_2/C_2H_5B_{10}H_{12}$ is the highest rated combination with a completely different oxidizer.

The same criteria used to rate the liquid bipropellant combinations were used to rate the hybrid systems according to the five areas of Table 35 . The results of these ratings are presented in Tables 51 to 55 . The area ratings were combined using the weighting factors of Table 35 , to obtain the overall hybrid rating presented in Table 56.

The metallic-additive systems have been treated in a different manner. In Table 57 ,the specific impulse is given for the bipropellant combination and for the additive systems. The percent by weight of additive is indicated in parentheses. Systems containing a large concentration of the metallic fuel (e.g., F_2/N_2H_4 + BeH_2) should not be considered as an additive system and thus were by-passed in preference to a hybrid system. Those combinations giving relative payload ratings higher than the 5.5 lower limit are tabulated in Table 58. The relative payload rating for bipropellant system is presented for comparison. Those additive systems giving less than a unit increment improvement above the bipropellant system were viewed as offering little advantage and thus were eliminated from contention.

Metallic additives to a fluorine-hydrogen system improve the theoretical specific impulse (Table 57), but because of the loss in bulk density of the propellant combination, the additive system provides less payload than can be realized with the bipropellant combination. A F_2/H_2 + Li system depicted in Fig. 11 shows the tradeoff between gain in specific impulse and the relative weights of the three propellants.

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TABLE 49

1975 BI-PROPELLANT COMBINATION LAUNCH OPERATION RATING

Propellant Combination	Rating	Propellant Combination	Rating
O_2/H_2	80	N_2F_4/B_2H_6	40
N_2F_4/N_2H_4	65	FLOX(90-10)/C_2H_6	35
N_2F_4/H_2	60	F_2/B_5H_9	30
F_2/MMH	55	OF_2/B_5H_9	30
F_2/N_2H_4	55	OF_2/Hydyne	30
F_2/UDMH	55	OF_2/MMH	30
F_2/50-50	55	OF_2/N_2H_4	30
F_2/Hydyne	55	OF_2/RP-1	30
F_2/Hydrazoid-P	55	OF_2/UDMH	30
FLOX(90-10)/MMH	55	OF_2/50-50	30
FLOX(90-10)/UDMH	55	N_2F_4/B_5H_9	30
FLOX(90-10)/50-50	55	ONF_3/B_5H_9	30
F_2/NH_3	45	OF_2/Hydrazoid-P	30
F_2/H_2	45	$OF_2/C_2H_5B_{10}H_{13}$	30
FLOX(90-10)/CH_4	45	OF_2/B_2H_6	20
F_2/CH_4	45	OF_2/CH_4	20
F_2/B_2H_6	40	OF_2/H_2	20
FLOX(30-70)/B_2H_6	40		

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TABLE 50

1975 BI-PROPELLANT COMBINATION OVERALL RATING

Propellant Combination	Rating	Propellant Combination	Rating
F_2/H_2	268	F_2/UDMH	197
F_2/N_2H_4	239	N_2F_4/N_2H_4	194
F_2/B_2H_6	232	OF_2/MMH	190
FLOX(90-10)/MMH	230	OF_2/50-50	190
F_2/NH_3	225	FLOX(90-10)/C_2H_6	186
FLOX(90-10)/50-50	224	OF_2/CH_4	186
F_2/Hydrazoid-P	223	OF_2/RP-1	185
F_2/B_5H_9	221	F_2/Hydyne	185
		OF_2/N_2H_4	184
$OF_2/C_2H_5B_{10}H_{12}$	215	N_2F_4/B_2H_6	184
F_2/50-50	214	OF_2/UDMH	183
OF_2/H_2	209	ONF_3/B_5H_9	181
F_2/MMH	208	OF_2/Hydyne	181
FLOX(90-10)/UDMH	207	F_2/CH_4	181
OF_2/B_5H_9	203	N_2F_4/B_5H_9	176
OF_2/B_2H_6	202	OF_2/Hydrazoid-P	176
FLOX(90-10)/CH_4	201	FLOX(30-70)/B_2H_6	171
N_2F_4/H_2	197	O_2/H_2	159

TABLE 51

1975 HYBRID PROPELLANT COMBINATION

PERFORMANCE RATING

Propellant Combination	Rating
F_2/BeH_2	97
F_2/Li	91
F_2/LiH	89
NF_3/BeH_2	82
F_2/AlH_3	81
$OF_2/LiBH_4$	70
N_2F_4/Li	67
$H_2O_2/(HBeBH_4)_2$	67
N_2H_4/Be	60
N_2O_4/Be	59
ClO_3F/BeH_2	58
O_2/BeH_2	56
NF_3/Li	55

TABLE 52

1975 HYBRID PROPELLANT COMBINATION

RELIABILITY RATING

Propellant Combination	Rating
F_2/AlH_3	73.5
F_2/BeH_2	73.5
F_2/Li	73.5
F_2/LiH	73.5
$OF_2/LiBH_4$	73.5
N_2F_4/Li	73.5
N_2H_4/Li	73.5
O_2/BeH_2	73.5
$H_2O_2/(HBeBH_4)_2$	73.5
H_2O_2/BeH_2	73.5
N_2O_4/Be	73.5
ClO_3F/BeH_2	73.5
NF_3/Li	73.5

TABLE 53

1975 HYBRID PROPELLANT COMBINATION

OPERATIONAL ASPECTS RATING

Propellant Combination	Rating
N_2O_4/Be	83.1
N_2F_4/Li	80.3
N_2H_4/Be	78.8
$H_2O_2/(HBeBH_4)_2$	76.1
H_2O_2/BeH_2	76.1
ClO_3F/BeH_2	72.4
F_2/AlH_3	70.6
F_2/BeH_2	70.6
F_2/Li	70.6
F_2/LiH	70.6
NF_3/BeH_2	70.4
NF_3/Li	70.4
$OF_2/LiBH_4$	69.0
O_2/BeH_2	65.6

TABLE 54

1975 HYBRID PROPELLANT COMBINATION

DEVELOPMENT RATING

Propellant Combination	Rating
F_2/AlH_3	60.0
N_2O_4/Be	52.0
N_2F_4/Li	46.0
F_2/BeH_2	40.0
F_2/Li	40.0
F_2/LiH_2	40.0
N_2H_4/Be	40.0
NF_3/BeH_2	40.0
NF_3/Li	40.0
$H_2O_2/(HBeBH_4)_2$	34.0
H_2O_2/BeH_2	34.0
$OF_2/LiBH_4$	28.0
O_2/BeH_2	28.0
ClO_3F/BeH_2	28.0

TABLE 55

1975 HYBRID PROPELLANT COMBINATION

LAUNCH OPERATION RATING

Propellant Combination	Rating
N_2H_4/Be	50
$H_2O_2/(HBeBH_4)_2$	50
H_2O_2/BeH_2	50
N_2O_4/Be	50
O_2/BeH_2	40
F_2/AlH_3	30
F_2/BeH_2	30
F_2/Li	30
F_2/LiH	30
$OF_2/LiBH_4$	30
N_2F_4/Li	30
ClO_3F/BeH_2	30
NF_3/BeH_2	30
NF_3/BeH_2	30

TABLE 56

1975 HYBRID PROPELLANT COMBINATION OVERALL RATING

Propellant Combination	Rating	Propellant Combination	Rating
F_2/BeH_2	248	$OF_2/LiBH_4$	195
F_2/Li	237	$H_2O_2/(HBeBH_4)_2$	195
F_2/LiH	233	N_2O_4/Be	192
F_2/AlH_3	230	N_2H_4/Be	189
NF_3/BeH_2	220	ClO_3F/BeH_2	176
N_2F_4/Li	201	NF_3/Li	172
H_2O_2/BeH_2	199	O_2/BeH_2	170

TABLE 57

1975 METALLIC ADDITIVE PROPELLANT COMBINATIONS

Propellant Combination	Bipropellant Performance+	Performance with Metallic Additive: (Percent by Weight of Additive in Fuel)			
		AlH_3	Be	BeH_2	Li
F_2/CH_4	344	*	345 (15)	*	*
F_2/H_2	410	**	437 (30)	437 (34)	431 (39)
F_2NH_3	359	-	362 (22)		373 (61)
$F_2N_2H_4$	363	**	-	376 (88)	*
F_2/MMH	346	-	-	363 (30)	-
H_2O_2/B_5H_9	308	*	315 (25)	-	324 (36)
H_2O_2/Hybaline B-3	306	-	-	350 (50)	-
H_2O_2/NH_3	271	319 (80)	326 (37)	352 (50)	277 (50)
H_2O_2/N_2H_4	282	-	317 (16)	327 (20)	-
N_2O_4/MMH	288	-	-	331 (33)	-
N_2O_4/NH_3	269	311 (76)	316 (31)	346 (50)	273 (42)
OF_2/N_2H_4	339	-	342 (14)	-	-
O_2/CH_4	311	324 (67)	-	359 (36)	**
O_2/H_2	391	397 (67)	457 (47)	457 (56)	404 ( )
O_2/NH_3	294	325 (51)	331 (17)	361 (55)	**
O_2/N_2H_4	313	330 (24)	337 (22)	362 (26)	**

* Highest Performance as hybrid.
**Highest Performance as a bipropellant.
\+ Chamber Pressure, psia = 1000; Expansion Area Ratio = Optimum Sea Level

TABLE 58

METALLIC ADDITIVE PROPELLANT COMBINATIONS -

RELATIVE PAYLOAD RATINGS

	Additive Relative Payload	Bipropellant Relative Payload	Elimination
O_2 / H_2 + BeH_2	9.8	8.1	
F_2 / N_2H_4 + BeH_2	9.8	8.8	x
O_2 / H_2 + Be	9.8	8.1	
F_2 / MMH + BeH_2	8.8	7.0	
F_2 / NH_3 + Li	8.6	8.0	x
F_2 / NH_3 + Be	8.4	8.0	x
O_2 / N_2H_4 + BeH_2	6.9	2.4	
OF_2 / N_2H_4 + Be	6.7	6.2	x
F_2 / CH_4 + Be	5.8	5.7	x
O_2 / NH_3 + BeH_2	5.7	-1.0	
F_2 / H_2 + Li	5.6	11.6	x
F_2 / H_2 + Be	5.5	11.6	x
F_2 / H_2 + BeH_2	5.5	11.6	x
O_2 / N_2H_4 + Be	5.5	2.4	
H_2O_2/ Hybaline B-3 + BeH_2	5.5	0.6	

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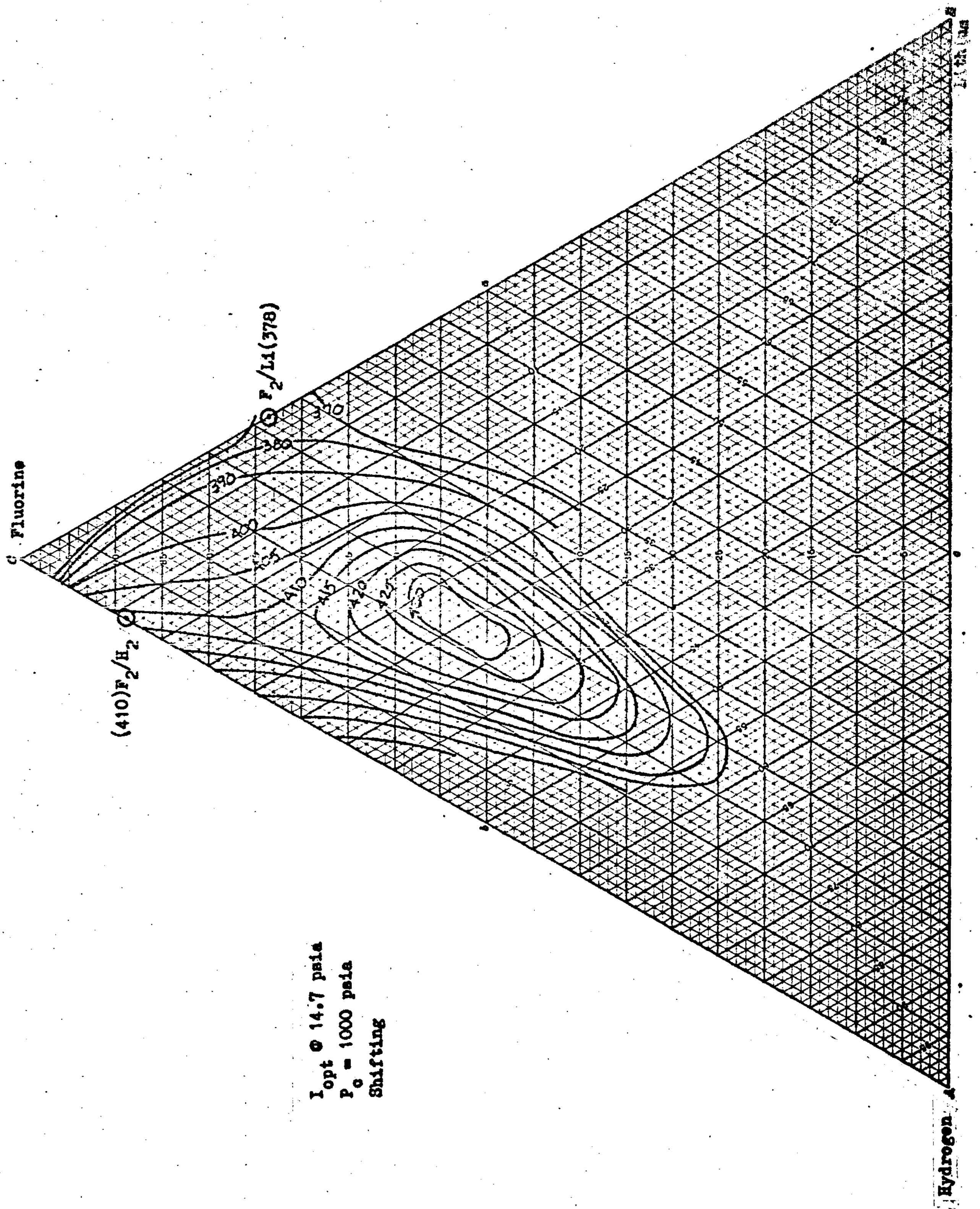

Fig. 11. Performance Contour for the $F_2/H_2/Li$ System

PROPELLANT-CANDIDATE SELECTION- 1975

The candidate propellants selected for 1975 are:

Best* 1970 bipropellant

F_2/H_2

$OF_2/C_2H_5B_{10}H_{13}$

F_2/BeH_2

$O_2/H_2 + Be$

$F_2/MMH + BeH_2$

These combinations include two of the high-rated bipropellant combinations and the best combination from 1970. The ratings of Table 50 indicate that the best 1970 combination rates very high for 1975. Also included in the selection are the highest rated hybrid and two of the highest rated metallic additive propellant combinations.

* Based on payload and system compatibility.

TABLE 59

ADVANCED APOLLO

1975 PROPELLANT COMBINATIONS

Propellant Combination	Weight Mixture Ratio (O/F)	Volume Mixture Ratio (O/F)	Sea Level (sec.)	Vacuum (sec.)	Bulk Specific Gravity	Combustion Temperature (F)	Hypergolic Ignition
F_2/N_2H_4	2.3	1.51	363	422	1.31	7550	Yes
F_2/H_2	8.0	0.37	410	474	0.46	6670	Yes
$OF_2/C_2H_5B_{10}H_{13}$	3.8	2.04	354	422	1.30	4990	Yes
F_2/BeH_2	5.0	-	(376)	440	1.24	8790	Yes
$O_2/H_2/Be$	0.85	-	457	533	0.25	4540	Yes
$F_2/MMH/BeH_2$	3.35	-	363	434	1.25	8290	Yes

APPENDIX A

TABLE 1A

PROPELLANT SURVEY - FUELS

I. Amine and CN Families

A. Hypothetical

B. Laboratory Characterization

C. Engineering Characterization

1. Ammonia
2. Hydrogen Cyanide
3. Methylamine
4. Acetonitrile
5. Ethylene amine
6. Dimethylamine
7. Ethylamine
8. Trimethylamine
9. Diethylamine
10. Aniline
11. Triethylamine
12. Dipropylamine
13. Diisopropylamine
14. Ethyl-n-Butylamine
15. n-Hexylamine
16. Cyanogen
17. Ethylene Diamine
18. Propylene Diamine
19. Dimethylamino Propylamine
20. N, N, N', N' Tetramethyl Propane-1.3 Diamine
21. N, N, N', N' Tetramethyl Butane-1.3 Diamine
22. 1.3 B13 (Dimethyamino) Propane
23. Diethylamino Propylamine
24. Tetramethyldiaminobutane
25. Diethylene Triamine
26. Tetramethyltetrazene
27. Tetracyanoethylene
28. Aminotetrazole
29. Diaminotetrazole
30. Triaminoguanidine
31. Triaminoguanadinium Azide
32. Triaminoguanadidium Hydrazinium Diazide

PROPELLANT SURVEY - FUELS

II. Hydrazines and Substituted Hydrazines

A. Hypothetical

B. Laboratory Characterization

1. Hydrazinium Nitrate
2. 1, 2 Ethylene Dihydrazine
3. Hydrazine Azide Hydrazinate
4. Diethylene Trihydrazine

C. Engineering Characterization

1. Hydrazine
2. Monomethylhydrazine
3. Unsymmetricaldimethylhydrazine
4. Symmetricaldimethylhydrazine
5. Trimethylhydrazine
6. Hydrazine Azide

III. Elemental Fuels

A. Hypothetical

B. Laboratory Characterization

C. Engineering Characterization

1. Hydrogen

PROPELLANT SURVEY - FUELS

IV. <u>Hydrocarbons</u>

A. Hypothetical

B. Laboratory Characterization

C. Engineering Characterization

1. Methane
2. Methylalcohol
3. Acetylene
4. Ethylene
5. Ethane
6. Ethylalcohol
7. Allene
8. Cyclopentane
9. Propane
10. Isopropyl Alcohol
11. Tetrahydro Furan
12. Butane
13. Pentane
14. Neopentane
15. Furfuryl Alcohol
16. Methyl Cyclopentane
17. Hexane
18. Trimethyl Botane
19. Benzene
20. Toluene
21. Heptene
22. Methylcyclohexans
23. Heptane
24. Octane
25. Diethylenecyclohexane
26. Diethylcyclohexane
27. Tetradecene
28. Tetradecane
29. $(CH_{1.9+})_x$ RP-1
    RP-3
    RP-4
    RP-5

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PROPELLANT SURVEY - FUELS

V. <u>Metallic Compounds</u>

A. Liquids

1. One component

a. Hypothetical

(1) Triborohydridoalane
(2) Aluminum Beryllohydride
(3) Hemoberyllium Borohydride
(4) Liquid Beryllium Polymers

b. Laboratory Characterization

(1) Triborohydridoaluminum Borohydride
(2) Tristriborohydride Alane

d. Engineering Characterization

(1) Aluminum Borohydride
(2) Monomethyl Aluminum Hydride
(3) Aluminum Trimethyl
(4) Aluminum Triethyl
(5) Diborane
(6) Pentaborane
(7) Propylpentaborane
(8) Ethyldecaborane
(9) Dimethylaminodiborane

2. Two Components-Coordinate

a. Hypothetical

(1) Zirconium Hybaline
(2) $H_2Be\ NH_2C = CNH_2 \cdot BeH_2$
(3) Amine Adducts of Diborane

b. Laboratory Characterization

(1) Hybaline B Series Adducts of $Be(BH_4)_2$

(a) B_{10}
(b) B_3

c. Engineering Characterization

(1) Hybaline A Series Adducts of $Al(BH_4)_3$

(a) A_1

(b) A_2

(c) A_4

(d) A_5

PROPELLANT SURVEY - FUELS

V. Metallic Compounds

B. Solids

1. One Component

a. Hypothetical

(1) Beryllium Aluminum Hydride

b. Laboratory Characterization

(1) Dilithium Aluminum Pentahydride
(2) Magnesium Aluminum Hydride
(3) Lithium Beryllium Hydride

c. Engineering Characterization

(1) Aluminum
(2) Aluminum Hydride
(3) Boron
(4) Decaborane
(5) Lithium Borohydride
(6) Sodium Borohydride
(7) Beryllium Borohydride
(8) Beryllium
(9) Berryllium Hydride
(10) Lithium
(11) Lithium Hydride
(12) Magnesium
(13) Magnesium Hydride

2. Two Component

a. Hypothetical

b. Laboratory Characterization

(1) Hybaline B Series   Adducts of $Be(BH_4)_2$

(a) $(NH_3)_5\ Be(BH_4)_2$
(b) $(N_2H_4)_2\ Be(BH_4)_2$
(c) $[(CH_3)_2\ N_2H_2]\ Be(BH_4)_2$
(d) $NH_2CH_2CH_2NH_2\text{-}Be(BH_4)_2$

c. Engineering Characterization

(1) Hybaline A Series
(a) $CH_3NH_2\text{-}Al(BH_4)_3$
(b) $H_2\ NCH_2CH_2NH_2\text{-}2Al(BH_4)_3$
(c) $H_2\ NCH_2C(CH_3)H\ NH_2\ \text{-}2Al(BH_4)_3$
(2) Tetramethylammarium Inboriohydride
(3) Dekazene

PROPELLANT SURVEY - FUELS

VI. Mixtures & Slurries

A. Amines

1. Hypothetical
2. Laboratory Characterization
3. Engineering Characterization
   a. MAF-1
   b. MAF-3
   c. MAF-4

B. Hydrazines and Substituted Hydrazines

1. Hypothetical
2. Laboratory Characterization
3. Engineering Characterization
   a. MHF-1
   b. MHF-2
   c. MHF-3
   d. MHF-4
   e. MHF-5
   f. 50-50
   g. Hydrazoid P
   h. CSC-2305
   i. CSC-2330

C. Hydrocarbons

1. Hypothetical
2. Laboratory Characterization
3. Engineering Characterization
   a. JP-x

D. Metallics

1. Hypothetical
   a. N_2H_4 + Be H_2
   b. N_2H_4 + L_1
   c. H_2 + Be
   d. H_2 + Al
   e. H_2 + L_1
   f. H_2 + B
   g. CH_4 + L_1
   h. CH_4 + Be
   i. CH_4 + BeH_2
   j. CH_4 + Al H_3
2. Laboratory Characterization
   a. N_2H_4 + Al H_3
   b. CH_3 N_2H_3 or $(CH_3)_2$ N_2H_2 + BeH_2
   c. N_2H_4 + B
   d. Al $(BH_4)_3$ + N_2H_4
   e. Mg $(BH_4)_2$ + N_2H_4
3. Engineering Characterization
   a. N_2H_4 + Na (BH_4)
   b. N_2H_4 + Al
   c. N_2H_4 + Be
   N_2H_4 + B

TABLE 2A

PROPELLANT SURVEY - OXIDIZERS

I. Halogen and Interhalogen Compounds
   a. Hypothetical
      1. Chlorine hexafluoride
      2. Bromine heptofluoride
      3. ClF_3O
      4. ClF_3O_2
      5. ClF_5O
   b. Laboratory characterization
      1. Chlorine pentafluoride
   c. Engineering Characterization
      1. Fluorine
      2. Bromine trifluoride
      3. Bromine pentafluoride
      4. Chlorine trifluoride
      5. Perchloryl fluoride

II. Oxygen, Peroxides, and Trioxides
   a. Hypothetical
   b. Laboratory Characterization
      1. Ozone
      2. tra (trifluorsmethy)
      3. Perfluoromethyl perfluoroethyl trioxide
      4. Tro (perfluoroethyl) trioxide
   c. Engineering Characterization
      1. Oxygen
      2. Hydrogen peroxide

PROPELLANT SURVEY - OXIDIZERS

III. Nitrogen Oxides & Nitro Compounds
- a. Hypothetical
- b. Laboratory Characterization
  1. NO_2F
  2. Dinitro difluoromethane
- c. Engineering Characterization
  1. Nitric Oxide
  2. Nitrogen Tetroxide
  3. Nitric Acid
  4. Tetranitromethane

IV. Oxygen Fluorides
- a. Hypothetical
  1. NF_2OF
  2. $(CF_3)_2$ NOF
  3. $(CF_3)_2$ N(O) OF
  4. NF_2CF $(OF)_2$
  5. $C(OF)_4$
- b. Laboratory Characterization
  1. O_2F_2
  2. O_3F_2
  3. O_4F_2
  4. Trifluoromethyl peroxyfluoride
  5. 2-Chlorotetrafluoroethyl oxyfluoride
  6. Perfluoroethyl oxyfluoride
  7. 2-Nitrotetrafluoroethyl oxyfluoride
  8. Perfluoroethyl peroxyfluoride
  9. Perfluoro isopropyloxyfluoride
  10. Perfluoroethyl fluoroxydifluoromethyl perioxide
  11. CF_3OO $C(CF_3)$ F OF
  12. Perfluoro t-butyloxyfluoride
  13. CF_3CF_2 OO $C(CF_3)F$ OF
  14. Difluoromethylene bis (oxyfluoride)
  15. Perfluoroethylidine bis (oxyfluoride)
  16. FO CF_2CF_2CFO
  17. Perfluoroisopropylidene bis (oxyfluoride)
  18. 1, 3 bis (fluoroxy) Perfluoropropane
  19. bis (1-fluoroxytetrafluoroethyl) peroxide
- c. Engineering Characterization
  1. OF_2

PROPELLANT SURVEY - OXIDIZERS

V. Nitrogen Fluorides

a. Hypothetical

1. ClF_2NF_2
2. $MClF_3NF_2$
3. $ClF(NF_2)_2$
4. $Cl(NF_2)_3$
5. $F_2N - CH_2C(NF_2)_2CH_2NF_2$

b. Laboratory Characterization

1. difluoramine
2. chlorodifluoramine
3. c-tromofluoroethyleneimine
4. difluoraminodibromofluoromelhane
5. chlorofluorodiazomethane
6. difluorominofluorodiazomethane
7. perfluoromethyleneimine
8. fluoroaminotrifluoromethane
9. difluoraminocyanicle
10. difluorodiazomethane
11. difluorominodifluoromethyleocyanate
12. trifluoromethyl (difluoroaminodifluoromethylamin
13. $(RF_3)_2N-NF_2$
14. methyl n, n- difluorocarpamate
15. difluoroeulfamyl fluoride
16. c-chloroformamidine
17. bis (difluoromind) chlorofluoromelhane
18. bis (difluoromind) dichloromethane
19. perfluoroformamidine (RFF)
20. bis (difluoramind) fluoromethane
21. bis (difluoramind) fluoromethylazide
22. bis (difluoramind) difluoromethane (H)
23. difluoramindfluoramind difluoromethane (HH)
24. bis (difluoramind) methane
25. trifluoroquanidine
26. methaxy bis (difluoramind) fluoromethane
27. bis (difluoramind) fluoroacetonitrile
28. bis (difluoramind) fluoromethylieocyanate
29. perfluoroamino methylcarbamyl fluoride
30. $CF_3N(NF_2)NFCF_3$
31. $C_2F_5N(NF_2)NFC_2F_5$
32. 1, 2 bis (difluoramind) ethane
33. perfluoro-3-amind-1-methyldiazinidine (E)
34. difluoramind difluoromethyl)-trifluoromethylfluoramine
35. perfluoro-3-amino-1-methyldiazinidine
36. methylpenfluorocarbamidate
37. $NF_2CH_2N(NO_2)CH_2NF_2$
38. $F_2N-C(CN)NF$ perfluoroquanyl cyanimide
39. 1, 3- dinitnato-2, 2-bis (difluoramind)propane
40. 1, 2 bis (difluoramind -2 methyl propane
41. perfluoroquanidine (PF6)
42. bis (difluoramind) fluoramine chloromethane
43. tris (difluoromind) fluoramethane (R)
44. aminobis (difluoramind) fluoramindmethane
45. tris (difluoramind) difluoromethraymelhane
46. perfluoro-N-methylquasnidine
47. perfluoro-3-amind-1-aminomelhyldiazinidine (E)
48. bis (difluoramind) trifheorometryl fluoramino-fluoro methane
49. tris (difluoramind) methoxy methane
50. methoxybis (difluoramind) fluoramind methane
51. tris (difluoramind) methyl isocyamate

PROPELLANT SURVEY - OXIDIZERS

52. perfluorotetra hydro-s-triazinone
53. 1, 2, 3 tris (difluoramind) propane
54. perfluorohoxahycho-s-triazine (M)
55. $(NF_2)_3COCH_2CH_2NH_2$
56. $(NF_2)_3COCH_2CH_2NH_2$
57. $(NF_2)_3CON{=}CHCOCCH_3$
58. $(NF_2)_3$ $C{-}OCH_2COOCH_3$
59. tetrakis (difluoramind)methane (T or)
60. perfluoro (aminomelhyl) quanidine
61. $F_{10}BG$
62. perfluoro-2-aminotetnahycho
63. 1, 2, 3, 4 tetrakis (difluoramino) propane
64. 2, 3, 4, 5 tetrakis (difluoramino) tetrahydrofuron
65. F_7B6
66. F_9 B6
67. F11 B6
68. perfluorotetrahydroammeline
69. bis tris (difluoromind) methoxy ethane
70. $NF_2CH_2{-}C(NF_2)H\ C(NF_2)HO\ C(NF_2)H\ C(NF_2)H\ CH_2NF_2$

c. Engineering Characterization
   1. NF_3 nitrogen trifluoride
   2. N_2F_4 tetrafluorohydrazine

VI. ONF Compounds
   a. Hypothetical
      1. NF_2ONF_2
      2. $FONF_2O$
      3. NF_2ONO
      4. NF_2ONO_2
      5. $CINF_2O$
      6. CIF_2NF_2O
      7. $F_2NO\ C(NF_2)_3$
      8. $MCIF_3ONF_2$
      9. NF_2OONF_2

PROPELLANT SURVEY - OXIDIZERS

VII. Oxidizer Mixtures

a. Hypothetical

1. $NF_3O + C(NF_2)_4$
2. $NF_3O + F_{11}BC$
3. $C(NO_2)_4 + N_2F_4$
4. $HNF_2 + C(NO_2)_4$
5. $HNF_2 + C(NO_2)_4 + N_2F_4$
6. $NF_3O + N_2F_4$
7. $NF_3O + ClF_3$
8. $ClF_3 + ClF_5$

b. Laboratory Characterization

1. $HNF_2 + N_2F_4$, 2. $N_2O_4 + ClF_3$

c. Engineering Characterization

1. $ClO_3F + ClF_3$
2. $ClF_3 + BnF_3$
3. $ClF_3 + ClO_3F + BrF_5$
4. $O_2 + O_3$
5. $O_3 + N_2O_4$
6. $O_2 + F_2$
7. MON $(N_2O_4 + NO)$
8. RFNA or MDFNA $(HNO_3 + N_2O_4)$
9. IRFNA $(HNO_3 + N_2O_4 + NF)$
10. MOXIE 1 $(ClF_3 + N_2F_4 + ClO_3F)$
11. WFNA
12. MOXIE 2 $(FC(NF_2)_3 + N_2F_4 + ClO_3F)$
    MOXIE 2a (46.5 w/o + 46.5 w/o + 7.0 w/o)
13. $N_2F_4 + ClO_3F$
14. $FC(NF_2)_3 + N_2F_4$
15. $FC(NF_2)_3 + N_2O_4$

TABLE 3A

SOLID OXIDIZERS

		Availability		
		Production	Development	Research
Ammonium Perchlorate	NH_4ClO_4	X		
Ammonium Nitrate	NH_4NO_3	X		
Nitronium Perchlorate	NO_2ClO_4		X	
Hydrazine Nitroform	$N_2H_5C(NO_2)_3$		X	
Hydrazine Perchlorate	$N_2H_5ClO_4$		X	
Hydrazine Diperchlorate	$N_2H_6(ClO_4)_2$		X	
Hydrazine Nitrate	$N_2H_5NO_3$		X	
Triaminoguanidinium Perchlorate	$(N_2H_3)_3C\ ClO_4$			X
Hydroxylamine Perchlorate	$HONH_3ClO_4$			X
1,2-bis(difluoramino) ethylene dinitramine	$O_2NNH\ CH(NF_2)CH(NF_2)NHNO_2$		X	
INFO 615	$(NF_2)_3CO\ NH_3\ ClO_4$			X
INFO 635	$(NF_2)_3CO\ CH_2NH_3ClO_4$			X
Lithium Perchlorate	$Li\ ClO_4$	X		

TABLE 4A

SOLID FUELS

		Availability		
		Production	Development	Research
Aluminum	Al	X		
Beryllium	Be	X		
Lithium	Li	X		
Aluminum Hydride	AlH_3		X	
Beryllium Hydride	BeH_2		X	
Boron	B	X		
Zirconium	Zr	X		
Dilithium Aluminum Pentahydride	Li_2AlH_5			X

TABLE 5A

POLYMERIC BINDERS

	Availability		
	Production	Development	Research
Hydrocarbon based			
carboxyterminated polybutadiene CTP	X		
polybutadiene acrylic acid PBAA	X		
polybutadiene acrylic acid acrylonitrile PBAN	X		
polyurethane PU	X		
polysulfide	X		
Oxidative (used with oxidative plasticizers)			
nitrocellulose NC	X		
nitroplasticized polyurethane	X		
polynitramine		X	
$-NF_2$ containing polymers			X
poly-bisdifluoraminopropylacrylate NFPA		X	
Low Carbon Polymers			
polyvinyltetrazole-triaminoguanidine			X
polytaz (pyrolysis product of triaminoguanidinium azide)			X
formaldehyde-hydrazine-UDMH			X
polyethylenehydrazine		X	

TABLE 6A

OXIDATIVE PLASTICIZERS AND ADDITIVES

	Availability		
	Production	Development	Research
Nitroglycerin NG	X		
trimethylolethane trinitrate TMETN	X		
triethyleneglycol dinitrate TEGDN	X		
diethyleneglycol dinitrate DEGDN	X		
bisdinitropropyl formal BDNPF	X		
bisdinitropropyl acetal BDNPA	X		
cyclotetramethylenetetranitramine HMX	X		
cyclotrimethylenetrinitramine RDX	X		
1,1,2,3-tetrakis difluoramino propane TDP			X
1,3-dinitrato 2,2-bis difluoramino propane DNBP			X
Similar $-NF_2$ or $-OF$ containing liquids			X

TABLE 7A

SOLID ADDITIVES (FUEL)

	Availability		
	Production	Development	Research
Triaminoguanidinium hydrazinium diazide THA		X	
Triaminoguanidinium azide TAZ		X	
Hydrazinium azide hydrazinate HAH		X	
Hydrazine bis borane HDB		X	
Dekazene			X
-hydrazino decaborane			X
Tetramethylaminonium triborohydride QMB3			X
Other solid hydrides			X

APPENDIX B

EFFECTIVE SERVICE MODULE VELOCITY INCREMENT

Propellant requirements for the service module may be estimated by combining the two mission types into an equivalent constant initial-gross-weight mission. The effective velocity increment for the equivalent mission is determined as follows:

$$W_{Ptotal} = W_1 \left[1\text{-exp} \left(- \Delta V_1 / gIs \right) \right] + W_2 \left[1\text{-exp} \left(\Delta V_2 / gIs \right) \right] \quad (1)$$

define Δ V effective by

$$W_{Ptotal} = W_1 \left[1\text{-exp} \left(\Delta V_{eff} / gIs \right) \right] \quad (2)$$

Assume:

I_s = 400 seconds, λP = 0.8, then

$$PL/W_2 = \frac{\lambda P - 1 + \exp\left(-\Delta V_2/gIs\right)}{\lambda P} = 0.63$$

$$W_2 = \frac{PL}{PL/W_2} = \frac{15,000}{0.63} = 24,000 \text{ pounds}$$

thus total propellant weight is

$$W_P = 90,000 \left[1\text{-exp} \left(-4460/400g \right) \right] + 23,000 \left[1\text{-exp} \left(-4300/400\ g \right) \right]$$

$$W_P = 90,000 \, (0.294) + 2300 \, (0.284) = 33,300 \text{ pounds}$$

The effective mission velocity increment of the service module is given by

$$1\text{-exp}\left(-\Delta V_{eff}/g400\right) = \frac{W_p}{90{,}000}$$

$$\Delta V_{eff} = 6000 \text{ fps}$$

APPENDIX C

VELOCITY INCREMENT EFFECTS ON VOLUME COMPARISON

From Table 14, it is seen that each propulsion system must deliver an ideal velocity increment which differs slightly from the typical case mentioned above. The effect of assuming a different mission velocity increment for the volume analysis may be demonstrated by an example. If an O_2/H_2 system is compared to the reference system, assuming an ideal velocity increment of 7000 fps

$$\frac{V_p}{V_{p\ ref}} = \frac{\left[1\text{-exp}\left(\frac{-7000}{g450}\right)\right]}{\left[1\text{-exp}\left(\frac{-7000}{g315}\right)\right]} \cdot \frac{\rho_{B\ -\ ref}}{\rho_B}$$

$$\frac{V_p}{V_{p\ ref}} = 0.769\ \frac{\rho_{B\text{-}ref}}{\rho_B} = 0.769\ f\ (\rho)$$

If the typical velocity increment is assumed to be 6000 fps

$$\frac{V_p}{V_{p\ ref}} = \frac{\left[1\text{-exp}\left(\frac{-6000}{g450}\right)\right]}{\left[1\text{-exp}\left(\frac{-6000}{g315}\right)\right]}\ f\ (\rho)$$

$$\frac{V_p}{V_{p\ -\ ref}} = 0.758\ f\ (\rho)$$

The difference in tank volume ratio resulting from the two ideal velocity increments is about 1.5 percent. Thus the volume comparison is not significantly affected by the velocity increment assumption.

APPENDIX D

APOLLO VEHICLE PROPELLANT-TANK VOLUME LIMITS

Before a candidate substitute-propellant combination is given detailed consideration for the 1970 Apollo propulsion systems, the feasibility of the propellant tanks fitting within the existing spacecraft structure should be evaluated. Propellant combinations requiring little component and structural rearrangement will be better suited as substitute propellants than those combinations which will require major spacecraft modifications to house the associated propellant tanks. The purpose of the study was to determine the maximum propellant tank volumes that can be carried within the existing spacecraft structures.

The Apollo stage designs that have been used as a basis for this investigation may be subject to change as the development of the system progresses. Major changes, invalidating the results presented are not likely to occur; thus, the trends shown should be valid. Since the results affect the propellant selection only as a limitation, small changes will have little consequence on the propellant-selection results. (For the finally selected propellant, detailed system design will be made in a later phase of the study.)

SERVICE MODULE PROPULSION SYSTEM (SPS)

Propellant storage for the SPS consists of two oxidizer and two fuel tanks located symmetrically about the longitudal axis of the Service Module between radial bulkheads as illustrated in Figs. 1D and 2D . The two remaining radial compartments, sectors I and IV of section D-D, Fig. 1D , house the LO_2 and LH_2 storage spheres in sector I, and three fuel cells, the power control relay box, the RCS Control Unit, and various technical equipment in sector IV. The oxidizer tanks are 51-inch I.D. cylindrical tanks with spherical ends, with a total length of 166 inches and total capacity of 352 cu ft. The two 45-inch I.D. cylindrical fuel tanks, 168 inches long with sperical ends, yield a fuel tank capacity of 282 cu ft. This propellant capacity results in about 16 percent ullage with the present, NTO/N_2H_4-UDMH(50-50), propellant combination.

Propellant-tank storage capacity can be improved by increasing the length or diameter of the existing design, and by addition of auxiliary tanks. The fuel and oxidizer tank lengths may be increased by 5 and 6 inches respectively with minor modifications to the forward bulkhead. Lengthening

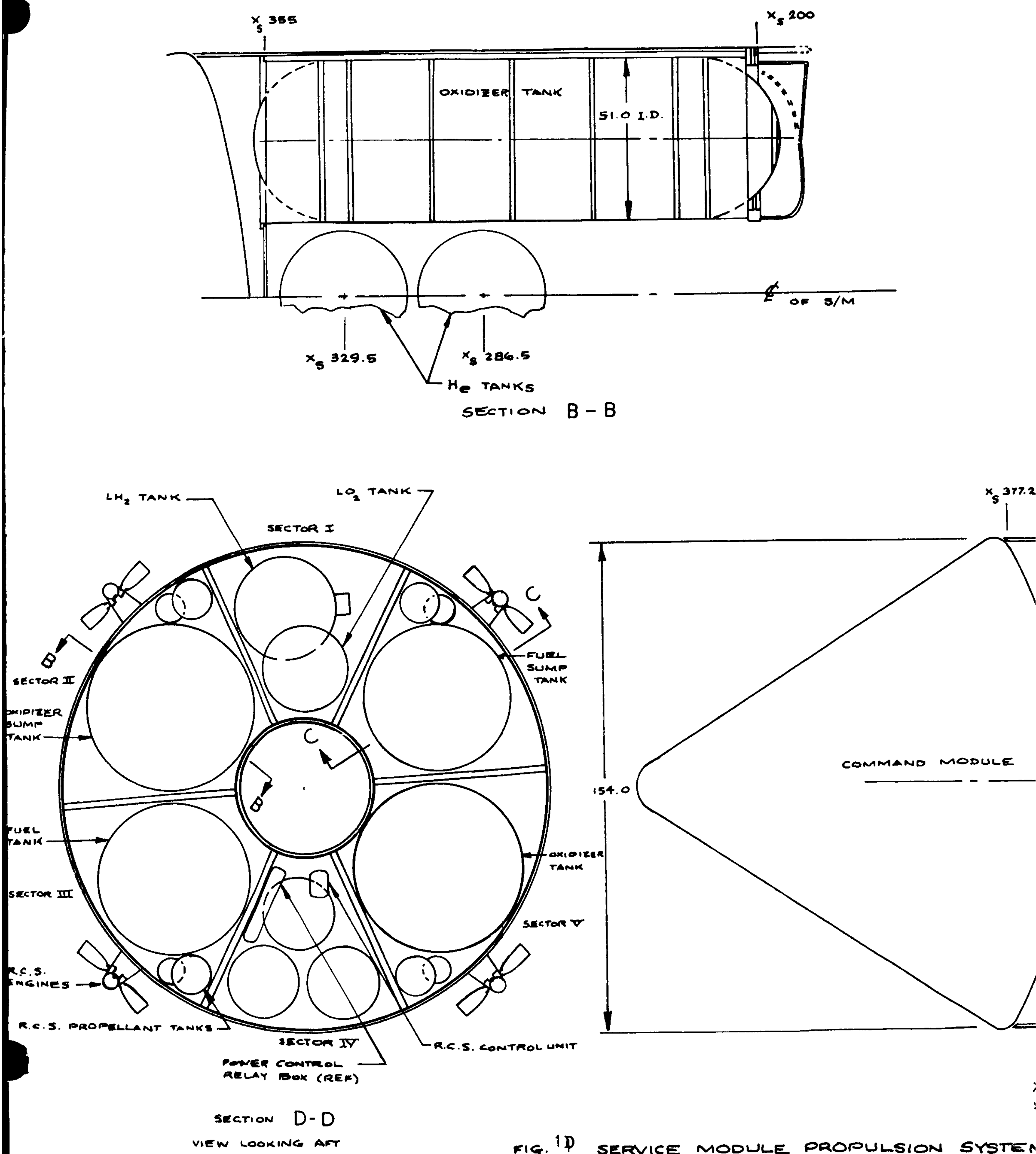
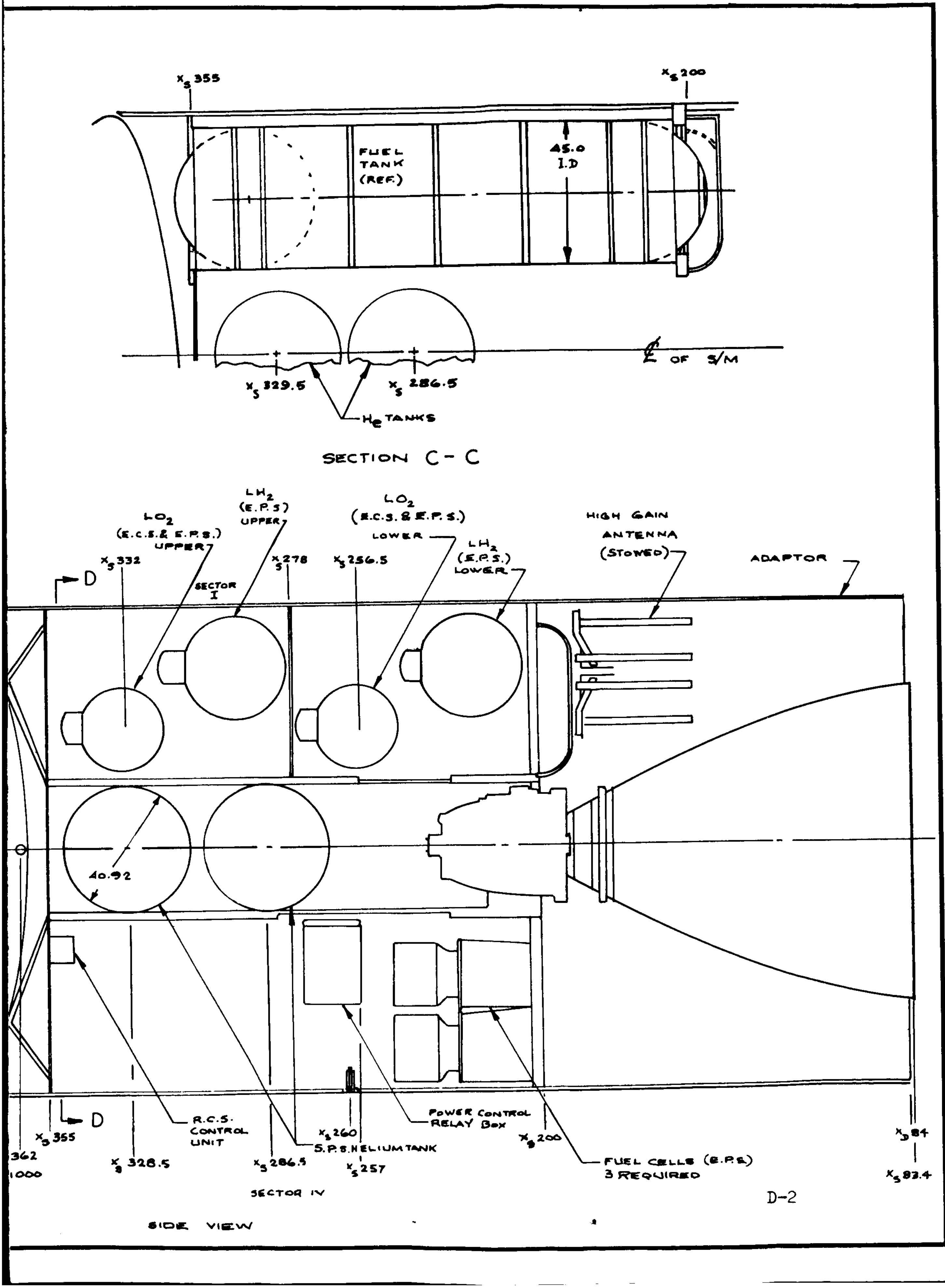


FIG. 1D SERVICE MODULE PROPULSION SYSTEM


X_S 355
X_S 200
FUEL TANK (REF.)
45.0 I.D
X_S 329.5
X_S 286.5
H_e TANKS
℄ OF S/M
SECTION C - C
LO_2 (E.C.S. & E.P.S.) UPPER
LH_2 (E.P.S) UPPER
LO_2 (E.C.S. & E.P.S.) LOWER
LH_2 (E.P.S.) LOWER
HIGH GAIN ANTENNA (STOWED)
ADAPTOR
X_S 332
SECTOR I
X_S 278
X_S 256.5
D
40.92
R.C.S. CONTROL UNIT
POWER CONTROL RELAY BOX
FUEL CELLS (E.P.S) 3 REQUIRED
S.P.S. HELIUM TANK
X_S 355
362
1000
X_S 328.5
X_S 286.5
X_S 260
X_S 257
X_S 200
X_S 84
X_S 83.4
SECTOR IV
SIDE VIEW

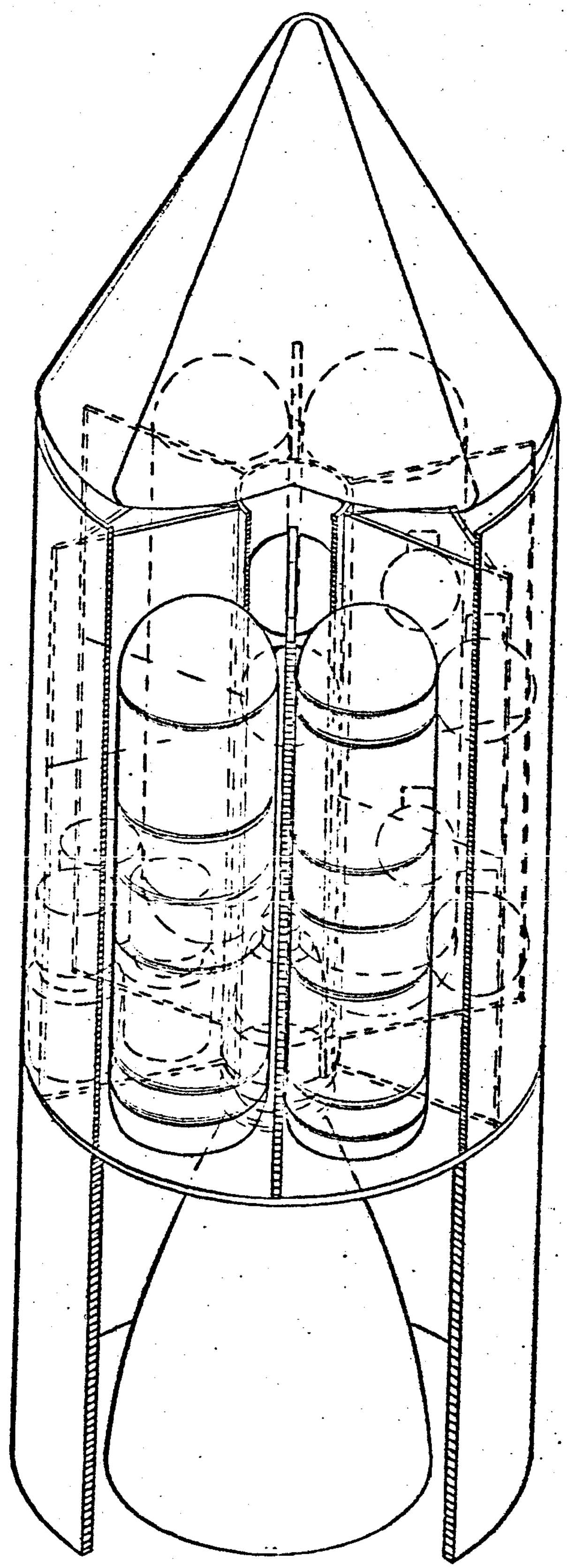


Fig. 2D Service Module Propellant Tank Arrangement the tanks through the rear bulkhead would require redesign of the high gain antenna mountings and thermal insulation from the engine; therefore, this method of increasing tank volume was considered impractical. The oxidizer tank diameter cannot be increased without redesign of adjoining bulkheads. From the detailed drawings (Ref. 9), it appears that the fuel tank diameter can be increased by 2 inches before bulkhead clearance becomes marginal.

Repackaging the LO_2 and LH_2 spheres (power source for the fuel cells) to allow relocation of the SPS helium tanks into the forward portion of sector I will permit addition of an auxiliary propellant tank in the center compartment above the engine. A 40-inch diameter cylindrical tank 100 inches in length with spherical ends can be placed in the region vacated by the helium spheres. However, increased pressurization system volume requirements, or LO_2 and LH_2 storage tank repackaging restrictions, may preclude relocation of the APS helium tanks.

Propellant-storage capacity resulting from the modifications discussed above are presented in Table 1D. The auxiliary tank volume is added to the larger oxidizer tank volume to indicate the maximum individual propellant volume attainable. The fuel and oxidizer tanks are interchangeable; therefore, the larger oxidizer volume illustrated can be considered either fuel or oxidizer storage capacity.

TABLE 1D

SERVICE MODULE PROPELLANT TANK VOLUME INCREASES

Tank Modification	Tank Volume, cu ft					
	Conservative			Optimistic		
	Oxidizer	Fuel	Total	Oxidizer	Fuel	Total
Increase Fuel Tank Diameter to 47 inches	352	289	641	352	289	641
Increase Tank Lengths, inches						
+6 Oxidizer Tank	+7			+7		
+5 Fuel Tank		+5	+12		+5	+12
Add Auxilliary Center Tank				+63		+63
Total Volume	359	294	653	422	294	716
Increase Over Present System,* cu ft	7	12	19	70	12	82
Increase, percent	2	4	3	20	4	13

*Present tank volumes: Oxidizer, cu ft = 352
Fuel, cu ft = 282

LEM DESCENT PROPULSION SYSTEM (DPS)

The DPS is arranged about a "cross" structure as shown in the simplified top view of Fig. 3D and the perspective drawing of Fig. 4D . Enclosed within the structure are 4 propellant tanks (2 oxidizer and 2 fuel tanks) arranged symmetrically about the vehicle vertical axis. Each tank is a cylinder 12 by 51 inches in diameter with spherical ends (total length, 63 inches) and has a capacity of 54.4 cu ft, for a total capacity of 217.6 cu ft. This volume results in about 4-percent ullage in the present system. A helium pressurization bottle and an oxygen tank are stored in two of the triangular spaces made by the arms of the support structure. The spaces are tapered shapes of triangular cross-section as illustrated in Figs. 5D and 6D . The lower one-third of the two triangular spaces shown without components are scientific-equipment storage areas as indicated in the figure. The landing engine is supported in the center of the cross-structure.

Increased tank capacity can be achieved by (1) enlarging the present tanks and (2) adding tanks in the triangular spaces. Enlarging the present tanks without redesign of the landing gear support structure appears marginal unless the tanks can be expanded into the engine compartment. The volume increases from various geometric changes are presented in Table 2D . The most likely tank modification appears to be a tank diameter increase to 55 inches, resulting in a 23-percent increase in tank volume.

Rearranging the helium tanks and oxygen storage tanks into the space above the scientific equipment storage area will allow use of the two remaining triangular spaces to store additional propellant. Auxiliary propellant tanks may be enclosed entirely within the triangular envelope or extended through the outside boundary of the space as shown in Fig. 6D . The two enclosed tanks result in 7 cu ft propellant storage capability while two of the larger tanks would add 34 cu ft of tank volume.

Tank volume increases based on conservative and optimistic changes in tank design are presented in Table 3D . The auxiliary tank volumes are added to the main fuel tank volumes to indicate the available increase in individual propellant volume.

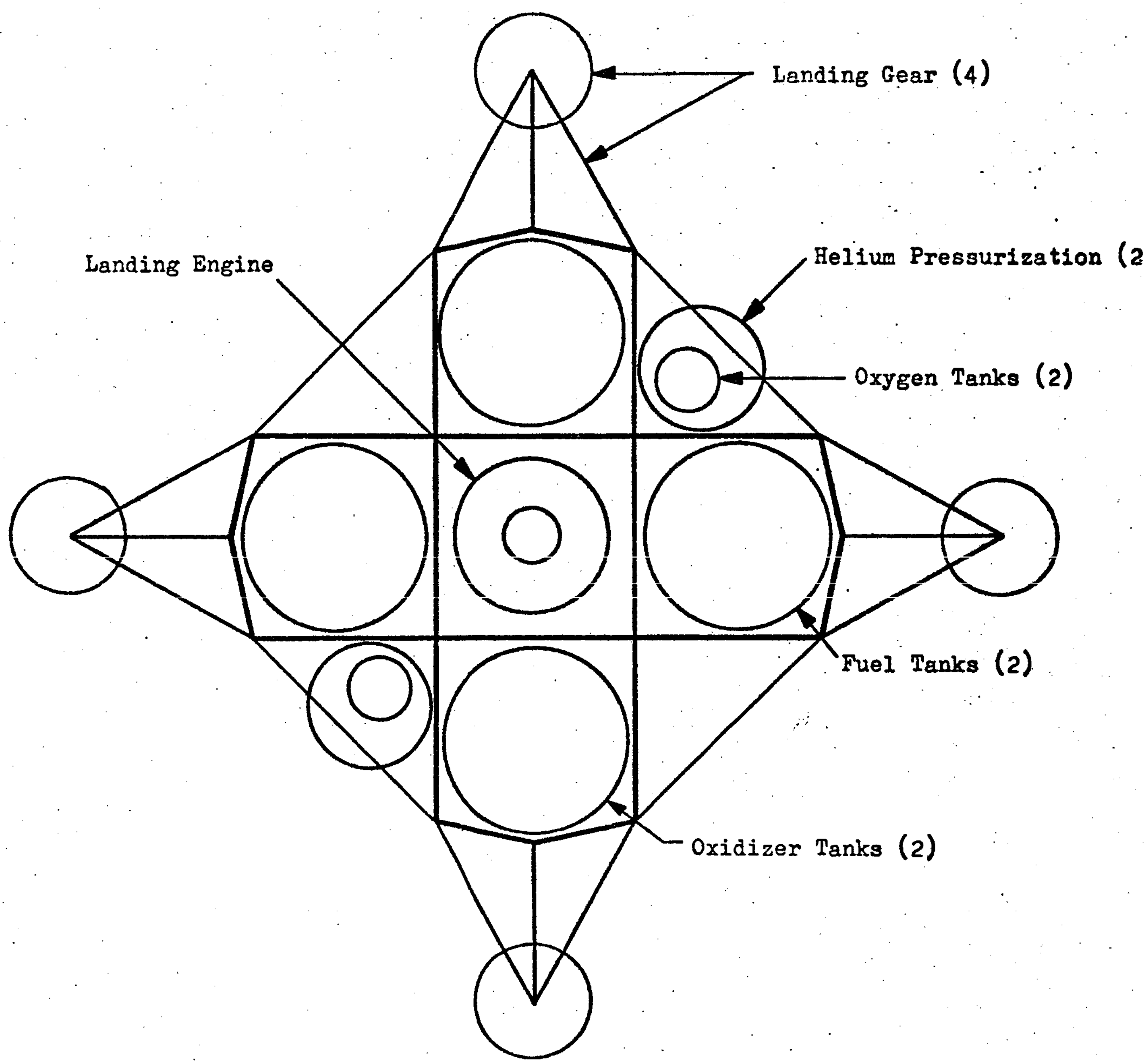


Fig. 3 D Simplified Top View of LEM Descent Propulsion System

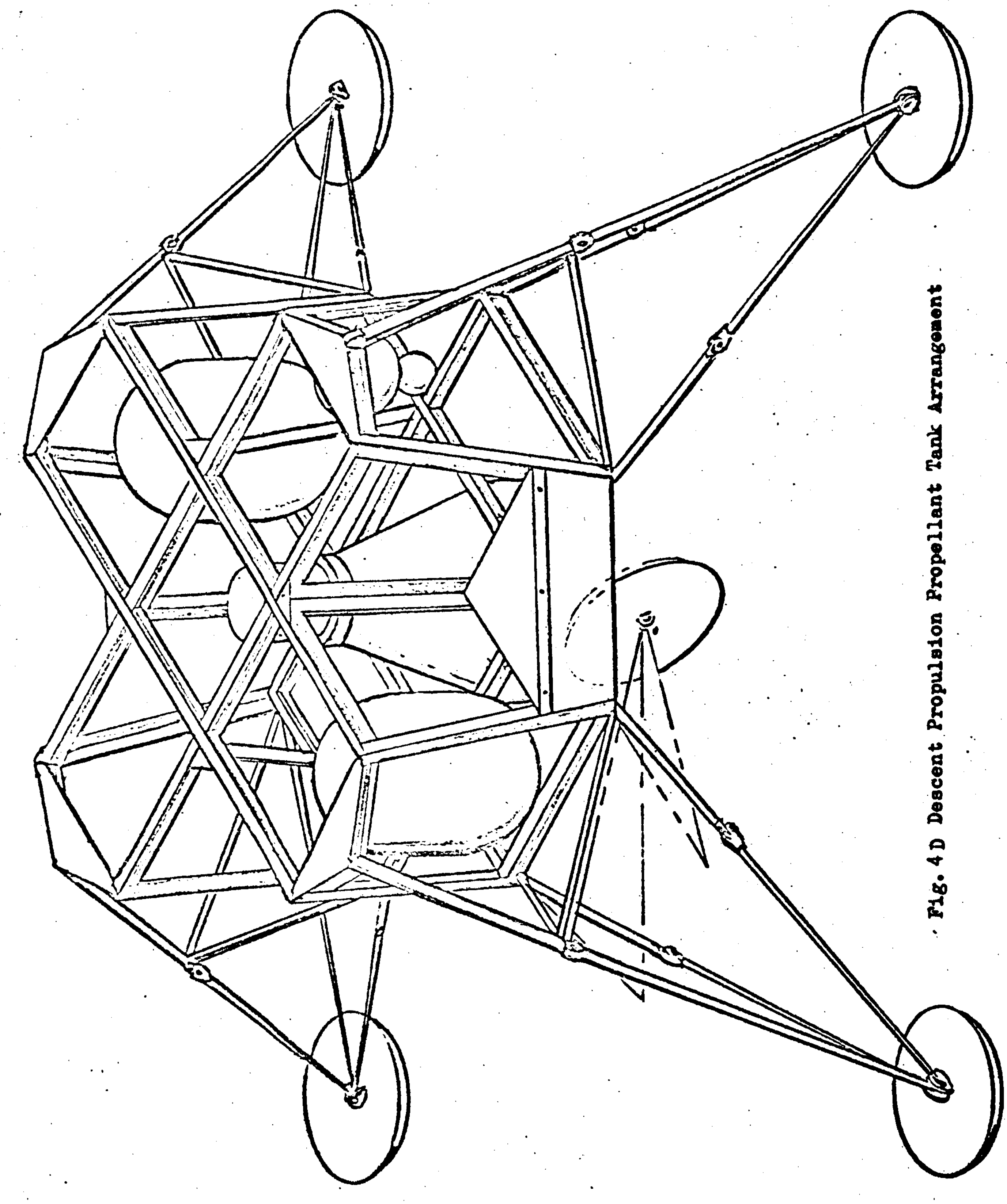


Fig. 4D Descent Propulsion Propellant Tank Arrangement

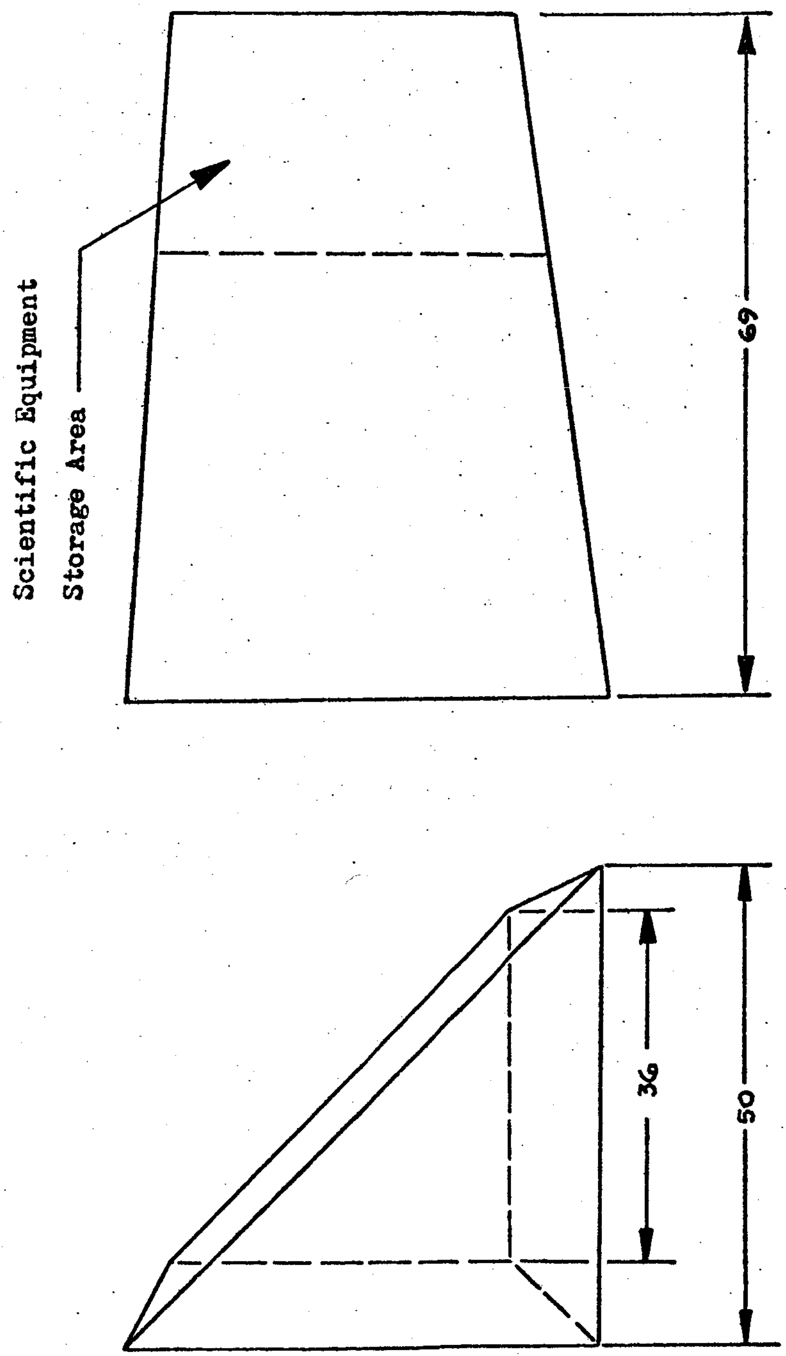


Fig.50 DPS Triangular Structure Dimensions

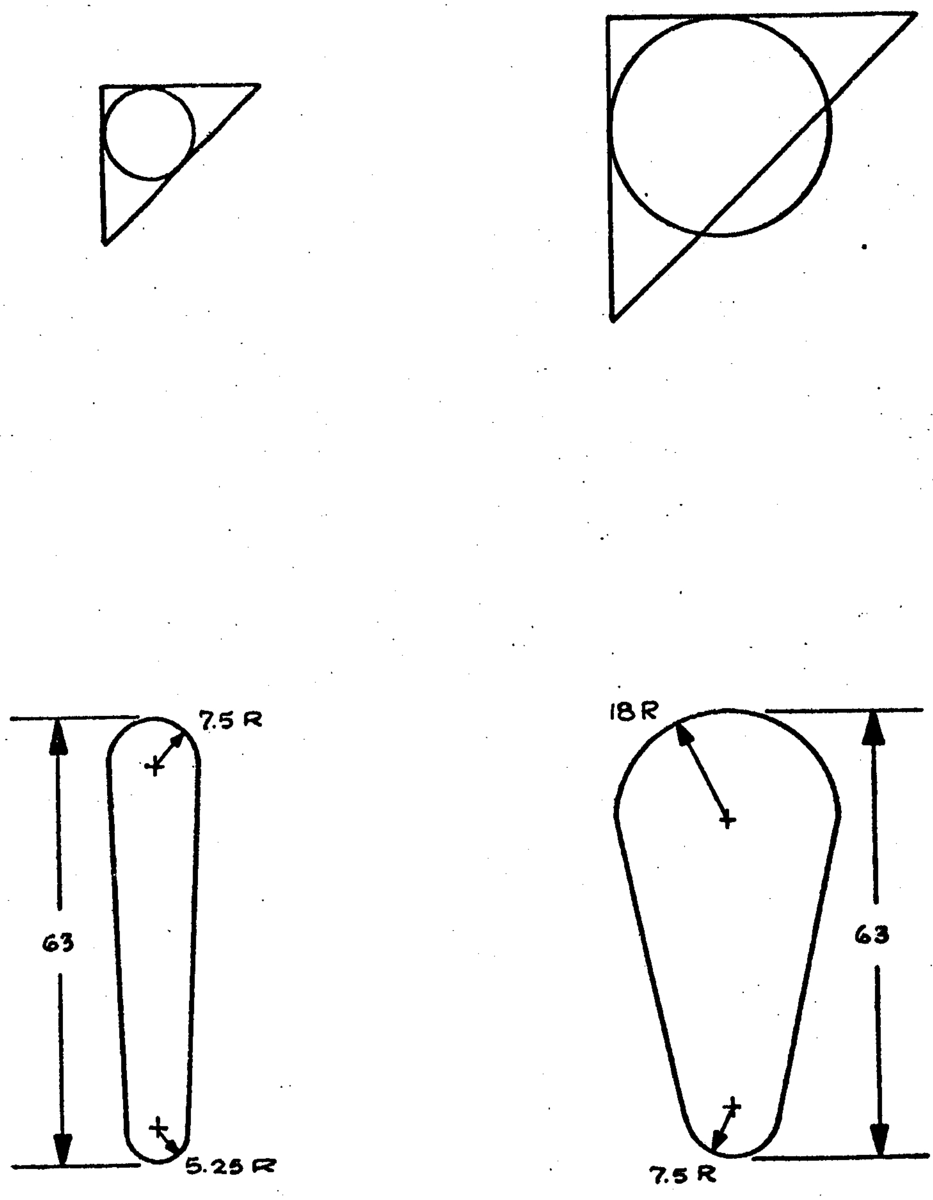


Fig.6D Tank Configurations For Triangular Spaces of
LEM Descent Stage (not to scale)

TABLE 2D

VOLUME INCREASE DUE TO GEOMETRIC MODIFICATION OF DESCENT STAGE TANKS

Tank Modification	Tank Volume, cu ft (each tank)	Volume Increase, percent
Present Design (volume) Diameter, 51 inches Length, 63 inches	54.4	0
Increase Diameter to		
a) 55 inches	67	23
b) 60 inches (length = 63 inches)	85	56
Substitute 2:1 Elliptical Ends Diameter, 51 inches Length, 63 inches	65	19
Increase Length to 68 inches Diameter, 51 inches Spherical ends	60	10

TABLE 3D

LEM DESCENT STAGE PROPELLANT TANK VOLUME INCREASES

Tank Modification	Tank Volume, cu ft					
	Conservative			Optimistic		
	Oxidizer	Fuel	Total	Oxidizer	Fuel	Total
Increase Existing Tank Diameter to:						
55 inches	134	134	268	170	170	340
60 inches						
Add Auxilliary Tanks:						
Enclosed		+7	+7			
Expanded					+34	+34
Total Volume	134	141	175	170	204	374
Increase Over Present System*						
cu ft	25	32	57	61	95	156
percent	23	29	26	56	87	72

*Present tank volumes: Oxidizer, cu ft = 109 (2 tanks)
Fuel, cu ft = 109 (2 tanks)

LEM ASCENT PROPULSION SYSTEM (APS)

The Ascent Propulsion System tank configuration is illustrated in Figs. 7D and 8D. One oxidizer and one fuel tank are contained in each of two propellant compartments located symmetrically about the longitudinal axis of the spacecraft. The present tanks are 40-inch spheres of 19.4 cu ft each, resulting in approximately 25-percent ullage in the present system. A 2.2 cu ft water storage tank and a 19-inch diameter spherical helium storage bottle are also stored in each compartment. Outboard of each oxidizer tank are the propellant and helium tanks for the Reaction Control System (RCS).

Increased tank capacity can be achieved by adding a 15-inch cylindrical section to the fuel (or oxidizer) tanks as shown in Fig. 9D . If the propellants are thermally compatible, a common bulkhead between the oxidizer and fuel tanks can be used to further increase the storage capacity as illustrated in Fig. 10D. Tank diameter cannot be increased without redesign of the vehicle surface structure because the present tank designs utilize the largest possible tank diameter (vis. section F-F of Fig. 8D) within the existing vehicle structure.

Substitution of 2:1 elliptical ends instead of spherical ends produces some added tank capacity to each tank. However, since the present vehicle structure above the fuel tank is spherical, addition of a fuel tank elliptical upper end will require a modification to the adjoining spacecraft structure. In each modification suggested above, the present location and arrangement of the RCS tanks areretained, but the water and SPS helium tanks are relocated.

Resultant tank volumes incorporating the tank redesign discussed above are presented in Table 4D . The larger tank is designated as the fuel tank to indicate the extent that individual propellant storage can be increased. Since the tanks have common boundaries, an exchange of tank capacity between the fuel and oxidizer tanks is possible, yielding an infinite range of propellant volume ratios within the total tank capacity indicated in the table. Therefore, the important value given by Table 4D is the total tank volume available.

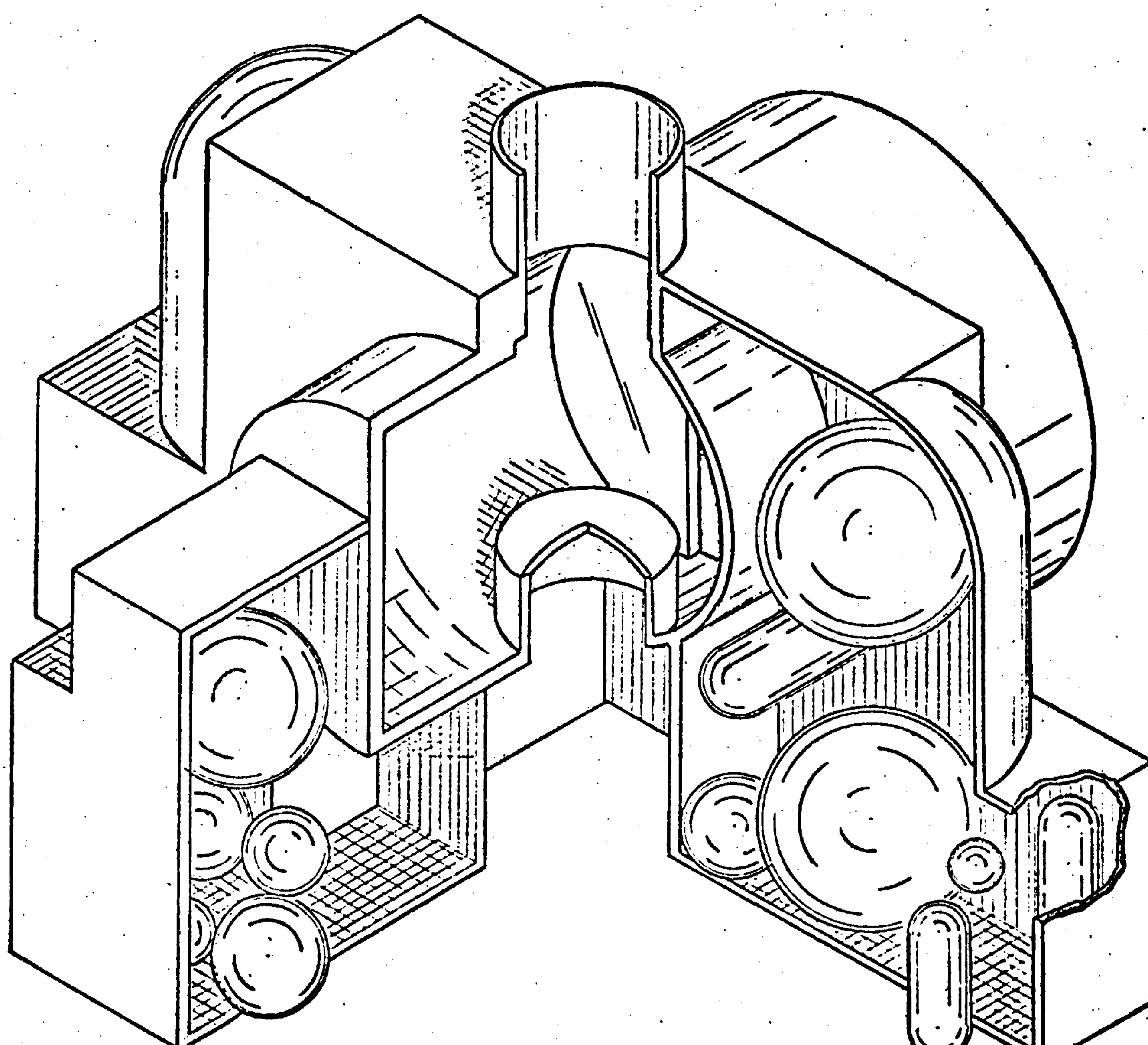
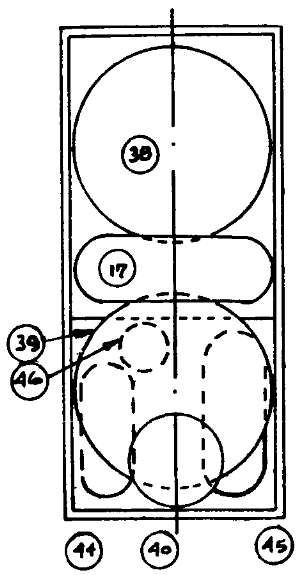
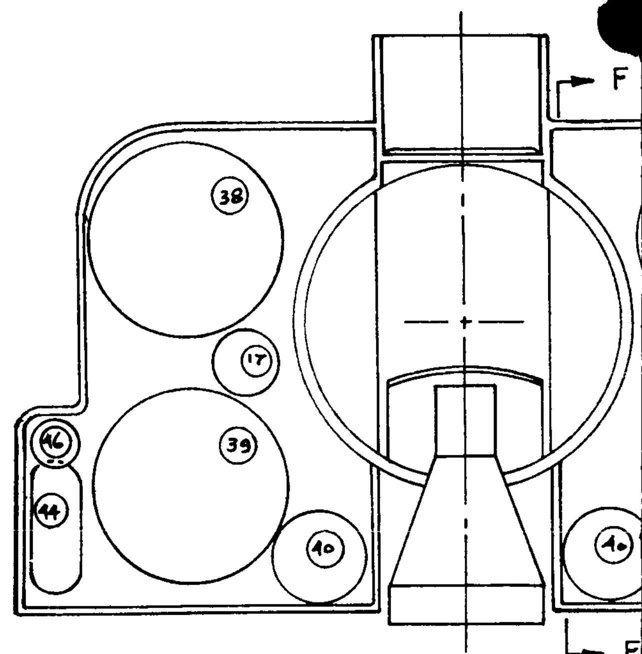


Fig. 7D Ascent Propulsion Propellant Tank Arrangement



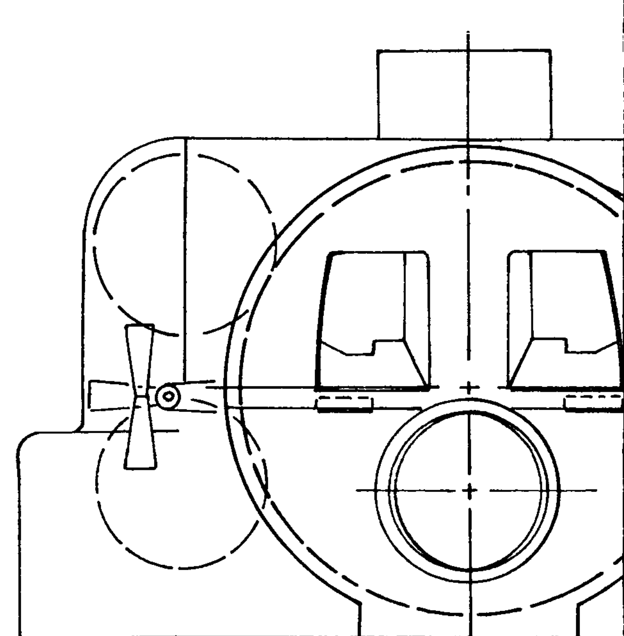
SECTION F-F



SECTION C-C

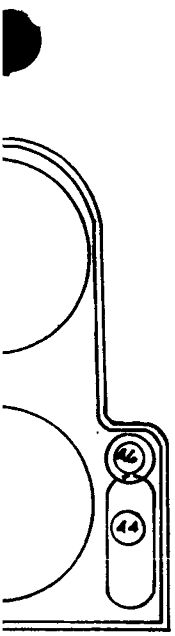
LEGEND

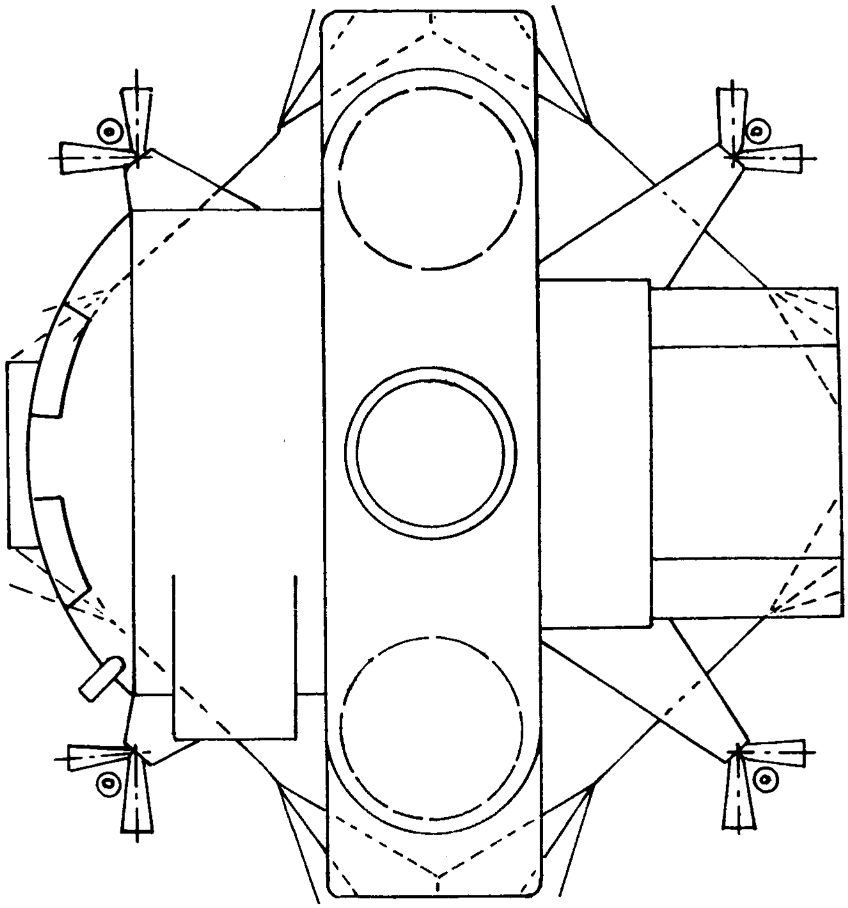
17	H_2O STOWAGE	(2)
38	FUEL TANK	(2)
39	OXIDIZER TANK	(2)
40	APS HELIUM	(2)
44	RCS FUEL	(2)
45	RCS OXIDIZER	(2)
46	RCS HELIUM	(2)



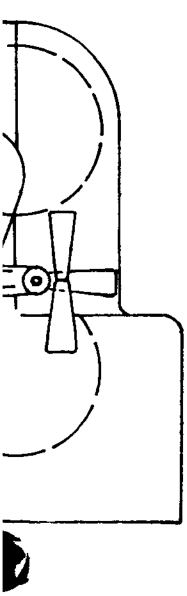
TOP VIEW

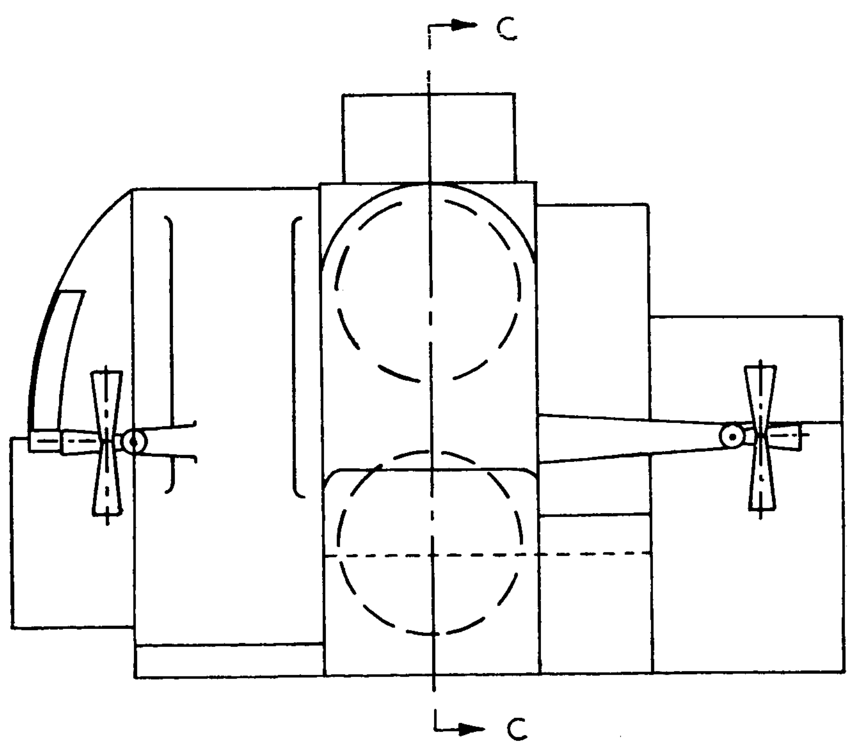
Fig. 8D ASCENT P





TOP VIEW





SIDE VIEW

SION SYSTEM TANK CONFIGURATION

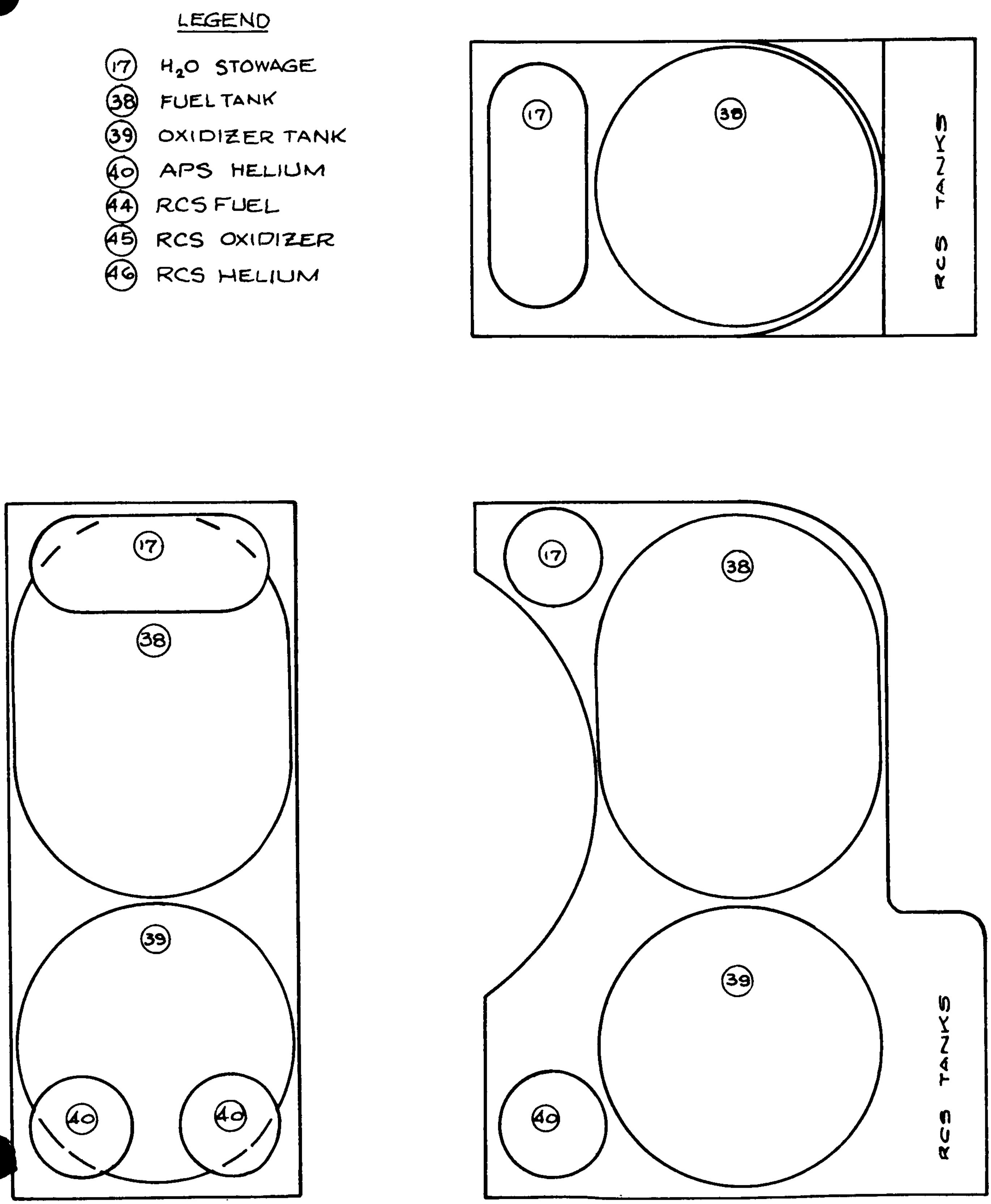


Fig. 9D Ascent Stage Expanded Propellant Tank Arrangement

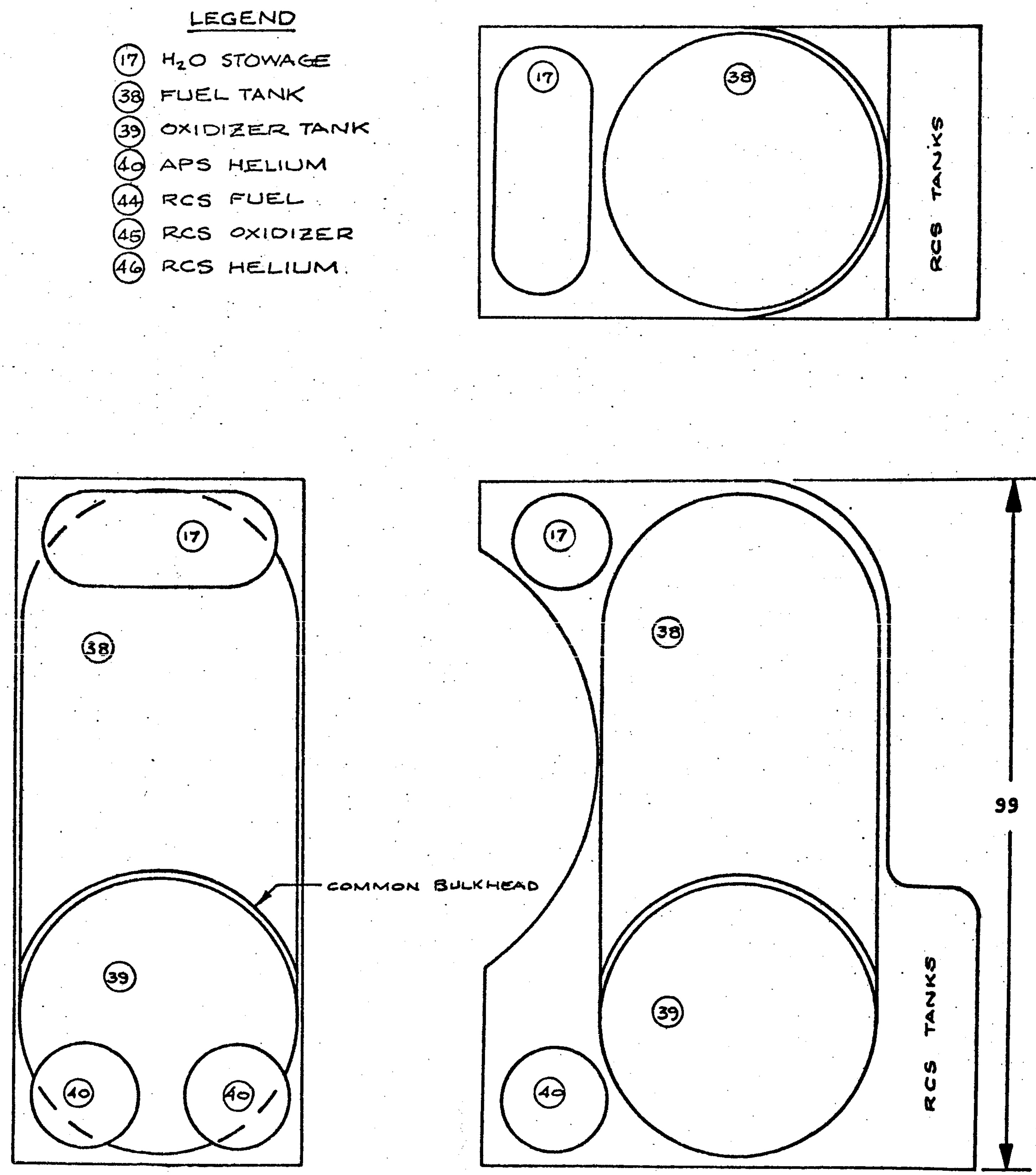


Fig. 10DAscent Stage Cylindrical Propellant Tanks, Common Bulkhead

TABLE 4D

LEM ASCENT STAGE PROPELLANT TANK VOLUME INCREASES

Tank Modification	Conservative			Optimistic		
	Oxidizer	Fuel	Total	Oxidizer	Fuel	Total
Add 15 inch section to fuel tank (Fig. 9D)	38.8	60.6	99.4			
Add cylindrical section and common bulkhead				38.8	81.4	120.2
Substitute 2:1 elliptical ends:						
excluding tops of fuel tanks	+4.8	+2.4	+7.2			
Total Volume, ft^3	43.6	63.0	106.6	41.2	83.8	125.0
Increase Over Present System:*						
ft^3	4.8	24.2	29	2.4	45	47.4
Percent	13	62	37	6	116	61

*Present tank volumes: Oxidizer, 38.8 cu ft (2 Tanks)
Fuel, 38.8 cu ft (2 Tanks)

Tank Volume Limits

Maximum attainable total and individual tank volumes resulting from the study for the Service Module propulsion system, SPS, the LEM Descent propulsion system, DPS, and the LEM Ascent propulsion system APS, are presented in Table 5D . The individual tanks are designated as oxidizer or fuel tanks, but the designations are interchangeable.

TABLE 5D

APOLLO SPACECRAFT PROPULSION SYSTEMS TANK VOLUME LIMITS

Propulsion System	SPS	DPS	APS
Oxidizer tank Capacity, cu ft			
Present	352	109	39
Maximum Attainable	422	170	41
Fuel Tank Capacity, cu ft			
Present	282	109	39
Maximum Attainable	294	204	84
Total Tank Capacity, cu ft			
Present	634	218	78
Maximum Attainable	716	374	125
Increase, percent	13	72	61

APPENDIX E

PROPELLANT THERMAL STORAGE IN SPACE

The Apollo mission covers an extended period of time in which the vehicle is exposed to the space environment of the earth-moon system. Therefore, thermal storage is one of the criteria affecting selection of propellant combinations for application in an advanced Apollo.

During the several days of the mission, propellants for the three propulsion systems must be thermally protected to prevent: (1) an excessive rise in tank pressure; (2) a propellant from freezing; (3) a large loss in propellant from boil-off. Attitude control of the vehicle can provide some protection during the mission. Insulation of propellant tanks provides the additional protection to prevent a propellant from undergoing a bulk temperature change greater than a predetermined allowable range. Protection by attitude control is an invariant between propellant combinations. Therefore, only insulation-weight variations between propellant combinations is investigated. This weight is used as a rating factor of the propellant combinations. It is merely indicative of the relative degree of difficulty in thermal storage between various propellant combinations.

GEOMETRICAL CONSIDERATION

In space, heat is transferred to or from a propellant tank by both conductive and radiative paths. Conductive flow paths exist in the supports to the tank and the propellant feed lines. Along these paths, heat flows between: oxidizer and fuel (unless a common liquidus storage temperature exists) internal heat sources such as electronic equipment and the propellants and the vehicle skin and the propellant.

An examination of the preliminary designs of the Apollo spacecraft indicates that the tanks are not an integral part of the vehicle shell. Therefore, radiation from external heat sources, e.g., emission of the sun, and albedo of the earth and moon, impinges upon the vehicle skin instead of the tank wall. Tanks are heated by irradiation of the vehicle skin and any internal heat sources which are not isolated by shadow shielding.

Each propellant combination under consideration for the advanced Apollo vehicle has its individual volume requirements. Some of the combinations are dense enough to be loaded into the Apollo tanks. Others require varying degrees of tank enlargement. An exact study of the thermal storage problem would require tank sizes and specific vehicle designs for each propellant combination. From these designs, conductive and radiative paths could be determined and the required insulation weight calculated. Such a detailed investigation would require more effort than that available for obtaining a thermal storage rating of the propellant combinations.

To bring the problem within a reasonable boundary, all radiative heat sources or sinks are assumed to combine to give a reference equilibrium temperature at the outer boundary of the insulation. The propellant does not change the reference temperature by heat transfer. In other words, the reference temperature is unaffected by any change in the temperature of the propellant. Heat transfer between propellants and the outside surface of the insulation is analyzed.

A comparison between propellant combinations rather than an absolute value of insulation weight is the desired goal of this investigation. For this objective, the exact configuration of the Apollo tanks is not required. Instead, any convenient tank configuration can be used if consistency is maintained between propellant combinations. Spherical tanks are used in this analysis.

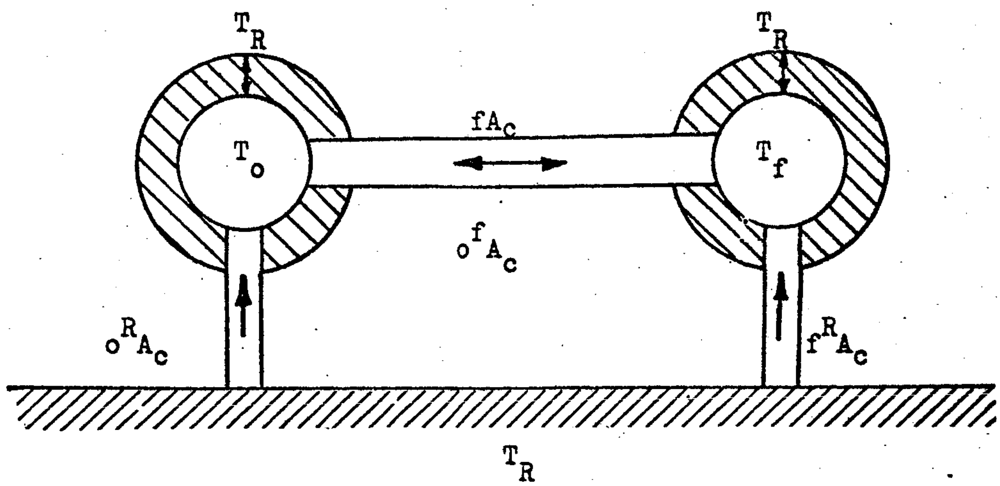
Many schemes of insulating spherical tanks could be devised. But, since this is a comparative analysis of various propellant combinations, selection of a best scheme to minimize the insulation weight is not required. A relative comparison of propellants based on a particular insulative scheme should agree with a similar comparison using some other technique of insulating. A layer of insulation is wrapped around the spherical tanks for this analysis.

HEAT TRANSFER

The assumption of a reference equilibrium temperature at the outer surface of the insulation and for the surrounding environment reduces the heat transfer to a conductive mode. Three conductive paths to the bulk of the propellant:

(1) Through supporting structure between tanks,
(2) Through supporting structure between tank and vehicle,
(3) Through the insulation,

are illustrated schematically in the following figure.



The total heat transferred to the propellant is:

$$Q_o = \frac{{}^f_o A_c \tau K_s (T_f - T_o)}{{}^f_o X_s} + \frac{{}^R_o A_c \tau k_s (T_R - T_o)}{{}^R_o X_s} + \frac{(A_o - {}^f_o A_c - {}^R_o A_c)\, \tau k_I (T_R - T_o)}{{}^R_o X_I} \tag{1}$$

$$Q_f = \frac{{}^o_f A_c \tau k_s (T_o - T_f)}{{}^o_f X_s} + \frac{{}^R_f A_c \tau k_s (T_R - T_f)}{{}^R_f X_s} + \frac{(A_f - {}^o_f A_c - {}^R_f A_c)\, \tau k_I (T_R - T_f)}{{}^R_f X_I} \tag{2}$$

It is difficult at present to ascertain what the reference temperature will be. In fact, the propellant temperature near the tank insulation will in all probability vary over a wide range between the time the vehicle is on the launch pad and the time for the final propulsion maneuver. This uncertainty leads to selection of a temperature band (-65F to + 160F) to include possible temperatures at the outer skin of the insulation. By definition, the reference temperature can be any value within the band.

The effect of this heat input on the propellant depends upon the storage system. For vented tanks, the heat input can raise the bulk temperature and vaporize propellant

$$Q_o = W_o (C_s)_o \; (t_o - T_o) + P_o \, W_o \, (\Delta h_v)_o \tag{3}$$

$$Q_F = W_f (C_s)_f (t_f - T_f) + P_f W_f (\Delta h_v)_f \qquad (4)$$

Cryogenics are generally tanked at their normal boiling temperature. With a venting system valve operating at a pressure equal to one atmosphere, all of the heat input to the cryogenic goes to vaporize part of the propellant. The first term of the equation is zero since there is no bulk temperature change. The second terms equated to equations 1 and 2 describe the thermal storage of a vented system.

NONVENTED TANKS

The present Apollo tanks are nonvented and are designed for noncryogenic propellants. Nonvented tanks are assumed for the advanced Apollo. The heat transfer lowers or raises the bulk temperature of the propellant.

$$Q_o = W_o (C_s)_o (t_o - T_o) \qquad (5)$$

$$Q_f = W_f (C_s)_f (t_f - T_f) \qquad (6)$$

Propellant is tanked at the lower of the two temperatures: + 68F or the normal boiling temperature. Heat transfer is reduced by insulation to prevent the propellant from reaching either of two bulk temperature limits: (1) a bulk temperature corresponding to a vapor pressure of 50 psia; (2) a bulk temperature corresponding to the normal freezing point.

The storage factor is computed using the larger of the two temperature differences between the propellant tankage temperature and the extremes of the reference temperature band. This is perhaps a pessimistic approach; however, it is consistant with the other assumptions in this comparison analysis of the storage requirements.

STEADY STATE

Combining the steady-state equations 1, 2, 5, and 6 gives an approximation to the effect of heat transfer between the propellant and the outer skin of the insulation. The specific heat capacity (c_p) is a function of the propellant temperature which in turn is a function of the time variable. Thus, a precise description requires evaluation of a definite integral between two time limits.

Such an evaluation would be impossible in this propellant comparison. As an alternative, the differential equations are solved as an approximation.

The areas of supports to the tank (${}^{f}_{o}A_c$, ${}^{r}_{o}A_c$) are very small with respect to the total tank surface area (Ao); therefore the difference between these areas is approximated by the total area. Equating 1 to 5 and 2 to 6, then combining terms gives:

$$W_o (C_s)_o \; (t_o - T_o) = \tau \left\{ \left[\frac{{}^{f}_{o}A_c}{{}^{f}_{o}X_s} \; k_s (T_f - T_o) \right] + \frac{A_o k_I}{{}^{R}_{o}X_I} \left[\left(\frac{{}^{R}_{o}A_c}{{}^{R}_{o}X_s} \right) \left(\frac{{}^{R}_{c}X_I}{A_o} \right) \left(\frac{k_s}{k_I} \right) + 1 \right] (T_r - T_o) \right\} \tag{7}$$

$$W_f (C_s)_f \; (t_f - T_f) = \left\{ \left[\frac{{}^{o}_{f}A_c}{{}^{o}_{f}X_s} \; k_s \; (T_o - T_f) \right] + \frac{A_f k_I}{{}^{R}_{f}X_I} \left[\left(\frac{{}^{R}_{f}A_c}{{}^{R}_{f}X_s} \right) \left(\frac{{}^{R}_{f}X_I}{k_I} \right) \left(\frac{K_s}{K_I} \right) + 1 \right] (T_R - T_f) \right\} \tag{8}$$

QUANTITATIVE COMPARISON

It is of interest to examine the bracketed portion of the second term of each equation. The three parenthetical factors of this term have typical values:

$$\frac{{}^{R}_{O}A_C}{{}^{R}_{O}X_s} \approx 10^{-1}$$

$$\frac{{}_{o}R_{x_I}}{A_o} \approx 10^{-4}$$

$$\frac{k_s}{k_I} \approx 10^{2}$$

The product of these three factors ($\approx 10^{-3}$) is small in comparison to unity, and therefore can be neglected in order to simplify the equations.

Two terms remain on the right-hand side of the equations. These two terms can be factored and with logic as above, one term is deleted because it is small compared to unity. The equations are reduced to:

$$W_o (C_s)_o \; (t_o - T_o) = \tau \left(\frac{A_o k_I}{{}_{o}R_{x_I}} \right) \; (T_R - T_o) \qquad (9)$$

$$W_f (C_s)_f \; (t_f - T_f) = \tau \left(\frac{A_f k_I}{{}_{f}R_{x_I}} \right) (T_R - T_f) \qquad (10)$$

The difference in storage requirement of various propellants is essentially described by these two equations which is the goal of the analysis. Since a relative comparison of the insulation weight rather than absolute values is the objective, the time factor (τ) of the mission is omitted and the equations restated to express the insulation thickness as a proportionality:

$$ {}_{o}R_{x_I} \propto \frac{A_o}{W_o} \cdot \frac{(T_R - T_o)}{(C_s)_o \; (t_o - T_o)} \qquad (11)$$

$$ {}_{f}R_{x_I} \propto \frac{A_f}{W_f} \cdot \frac{(T_R - T_f)}{(C_s)_f (t_f - T_f)} \qquad (12)$$

The propellant properties enter into these equations. These contributions are listed in Table 1E . As previously stated, the propellant tanks are assumed to be spherical in shape. The insulation material has the same physical shape. The equation for the weight of the insulation is the product of the thickness (Eq. 11 or 12), the surface area of the spherical tank, and the density of the insulation material. By insulating all propellant tanks with the same material, the density term can be omitted since it is a constant multiplier for all propellant combinations. The surface-area term cannot be dropped because the tank volume varies with the propellant combination. Thus, the weight equation of the insulation can be written:

$$ {}_{o}W_{I} \propto \frac{A_o^2}{W_o} \cdot \frac{(T_R - T_o)}{(C_s)_o (t_o - T_o)} \tag{13} $$

$$ {}_{f}W_{I} \propto \frac{A_f^2}{W_f} \cdot \frac{(T_R - T_f)}{(C_s)_f (t_f - T_f)} \tag{14} $$

The area of a spherical tank can be expressed in terms of a propellant density and the propellant weight contained therein. The fuel and oxidizer weights can be defined as functions of the weight mixture ratio, the specific impulse of the propellant combination, the ideal velocity increment, and the initial gross weight of the vehicle. Substituting the new parameters for the surface area and propellant weights in equations 13 and 14, dropping all constnat terms, and omitting the gross weight of the vehicle as invariant with propellant combinations, the insulation weight can now be expressed:

$$ {}_{o}W_{I} \propto \left[\frac{1}{\rho_o^2} \sqrt{\frac{MR\,(1-e^{-y})}{(1+MR)}} \right]^{2/3} \cdot \frac{(T_R - T_o)}{(C_s)_o (t_o - T_o)} \tag{15} $$

TABLE 1E

PHYSICAL PROPERTY CONTRIBUTION TO THERMAL STORAGE

Oxidizer	T_R	t_o	$\frac{(T_R - To)}{(C_s)_o\ (to-To)}$
BrF_5	- 65	- 65	4.75
ClF_3	+160	+112	5.94
ClF_5	+160	+ 58	9.66
ClO_3F	+160	+ 2	17.3
FLOX (30-70)	+160	-274	45.8
FLOX (90-10)	+160	-279	47.0
FNO_2	+160	- 35	12.6
F_2	+160	-280	47.3
HNO_3	- 65	- 43	2.82
H_2O_2 (98)	- 65	28	15.3
IRFNA	- 65	- 57	4.58
MDFNA	- 65	- 35	3.01
MON (75-25)	+160	+ 67	7.75
MON (85-15)	+160	+105	5.04
MOXIE 2	+160	- 42	11.6
NF_3	+160	-173	56.7
N_2F_4	+160	- 47	9.28

TABLE IE (CONT'D)

Oxidizer	T_R	t_o	$\frac{(T_R - To)}{(C_s)_o\ (to-To)}$
N_2O_4	- 65	+ 12	6.44
OF_2	+160	-197	43.4
ONF_3	+160	- 75	14.2
O_2	+160	-273	45.2
O_3	+160	-133	24.9
RFNA	- 65	- 56	2.56
B_2H_6	+160	- 90	10.0
B_5H_9	- 65	- 53	1.93
CH_4	+160	-225	14.7
C_2H_5OH	- 65	- 65	1.74
C_2H_6	+160	- 78	9.76
$C_3H_7NO_3$	- 65	- 65	2.41
$C_{10}H_{20}$	- 65	- 65	2.15
H_2	+160	-414	30.8
Hybaline A-5	- 65	- 58	1.70
Hydrazoid-P	- 65	- 65	1.57
JPX	- 65	- 65	1.68
Hydyne	- 65	- 65	1.77
MMH	- 65	- 65	1.46
NH_3	+160	+ 22	3.51

TABLE 1E(CONT'D)

Oxidizer	T_R	t_o	$\frac{(T_R - To)}{(C_s)_o (to-To)}$
N_2H_4	- 65	+ 35	5.48
N_2H_4 - UDMH (50-50)	- 65	+ 19	3.92
RP-1	- 65	- 55	2.41
UDMH	- 65	- 65	1.82
$C_2H_5B_{10}H_{13}$	- 65	- 65	2.00

$$ {}_{f}W_{I} \propto \left[\frac{1}{f^2} \quad \frac{(1-e^{-y})}{(1 + MR)} \right]^{2/3} \cdot \frac{(T_R - T_f)}{(C_s)_f \, (t_o - T_f)} \qquad (16) $$

where

$$ Y = \frac{\Delta V_I}{I_s g_o} $$

Equations 15 and 16 are the criteria for the evaluation and comparison of various propellant combinations. The calculated values are proportional to the weight of insulating material that would be required for a mission. A small value of W_I is indicative of an easily stored propellant. Summing the values for the oxidizer and the fuel gives an insulation factor comparison.

APPENDIX F

COOLING JACKET

THRUST CHAMBER PRESSURE DROP FACTOR

The cooling jacket pressure drop can be computed from the following relationship:

$$P = \frac{1}{2} \rho V^2 f \frac{1}{d}$$

assuming the friction factor (f) and the 1/d ratio constant or independent of the coolant and eliminating these factors from the equation.

$$\Delta P \propto \rho V^2$$

Now to develop this relationship in terms of the incident heat flux and liquid coolant properties and temperature limitations, a semi-emperical equation is used to describe the convective heat transfer coefficient for the coolant film.

$$N_u = \frac{h_c D}{K} = \phi Z_1 N_{Re}^{Z_2} \, N_{pr}^{Z_3} \left(\frac{T_B}{T_{wc}}\right)^{Z_4}$$

and

$$N_{Re} = \frac{\rho V D}{\mu} \quad \text{and} \quad N_{pr} \quad \frac{\mu C_p}{K}$$

and rearranging:

$$h_c = \phi Z_1 \frac{K^{1-Z_3} \, C_p^{Z_3} \, \mu^{Z_3 - Z_2}}{D^{1-Z_3}} \quad (\rho V)^{Z_2} \quad \left(\frac{T_B}{T_{wc}}\right)^{Z_4}$$

solving for ρV

$$\rho V = \left[\frac{h_c \quad {}^{Z_2-Z_3} \; D^{1-Z_2}}{\phi \, Z, \, K^{1-Z_3} \; Cp^{\,Z_3}} \qquad \left(\frac{Twc}{T_B} \right)^{Z_4} \right]^{1/Z_2}$$

The required coolant-side film coefficient h_c can be written in terms of the expected heat flux between the tube wall and the coolant and the associated temperature difference

$$h_c = \frac{Q/A}{T_{wc} - T_B}$$

By manipulation of known relationships involving the remaining parameters and substituting into the developed proportionality for the jacket pressure drop

$$\Delta P \propto \frac{\left[\frac{Q/A}{(Twc - T_B)} \qquad \frac{N_{pr}^{\;1-Z_3} \; D^{1-Z_2}}{\phi \, Z, \; \mu^{1-Z_2} \, Cp} \right]^{Z/Z_2}}{\rho}$$

In the above expression, ϕ is an entrance correction factor and Z is dependent on the roughness of the tube, therefore these factors along with the tube diameter may be considered common for all propellant combinations and eliminated from the equation. Also be assuming the Prandtl number and viscosity are similar for the various fuels, these two parameters are also excluded. In addition, the Prandtl number and viscosity enter the equation to exponents of 0.2 & 0.6 respectively ($Z_2 = 0.8$ and $Z_3 = 0.4$) which further weakens their influence in determining relative values for system pressure drops. The resulting relationship which is used in the overall comparison becomes:

$$\Delta P \propto \frac{\left[\frac{Q/A}{(Twc - T_B) \; Cp} \right]^{2.5}}{\rho}$$

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